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FAILSAFE/SAFE-LIFE INTERFACE CRITERIA

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Summa Corporation

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The purpose of this study was to develop rational failsafe/safe-life interface criteria for the design of helicopter structural components. Present specifications containing design criteria were reviewed and deficient areas defined. Rational design criteria were developed and are titled "Damage-Tolerant Design Criteria".

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The study involved a review of the current failsafe/safe-life design practices to analyze the appropriate specifications. This was accomplished by a review of pertinent published literature as well as visits to military helicopter manufacturers and appropriate Government agencies.

Damage-tolerant design concepts are presented for various helicopter structural components. They include redundant structure, controlled fracture structure and failure indicating systems. The Hughes OH-6A main rotor blade was selected as the structural component to establish the effects on life-cycle costs in designing to different structural criteria and using different damage-tolerant design concepts.

The study showed that increased emphasis should be placed on the "damaged" strength of structural components. The most costly item in designing to damage-tolerant criteria is the showing of formal compliance. Incorporating damage-tolerant techniques initially increases costs and weight; however, the components are expected to survive longer in service, thereby resulting in a lower life-cycle cost.

EUSTIS DIRECTORATE POSITION STATEMENT

This report is considered to provide a good review and analysis of failsafe/safe-life concepts. Results of the program indicate that additional work in the area of structural failure mechanisms and the application of fracture mechanics to fail-safe design would be fruitful.

This report has been reviewed by this Directorate and is considered to be technically sound. The technical monitor for this contract was Mr. A. J. Gustafson of the Structures Technical Area, Technology Applications Division.

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PREFACE

This final report is submitted as the required documentation pursuant to Contract DAAJ02-74-C-0005, Failsafe/Safe-Life Interface Criteria Study, between U.S. Army Air Mobility Research and Development Laboratory (USAAMRDL) of Fort Eustis, Virginia, and Hughes Helicopters (HH). The study presented herein was conducted at the HH facility, Culver City, California, from October 1973 through July 1974. The primary objective of the study was to identify deficient areas in current design practice and current Army specifications, as related to failsafe/safe-life design criteria, and to recommend corrections. The study also includes cost and weight trade-offs for recommended design criteria for various helicopter components. The effort was performed in three phases.

The author acknowledges the contributions made by Messrs. H. T. Lund, R. E. Moore, H. G. Smith, T. G. Summers, and R. A. Wagner of Hughes Helicopters.

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INTRODUCTION

A study has been conducted by Hughes Helicopters to develop rational criteria for the failsafe/safe-life design of helicopter component structures. The work was performed in three phases.

In Phase I, the state of the art of failsafe/safe-life design was determined with the objective of finding a relevant failsafe/safe-life design methodology for application to helicopter component structures. Pertinent published literature were studied, the latest trends of both commercial and military fixed-wing airframe manufacturers in regard to failsafe/safe-life design were determined, and current research and testing by various agencies were reviewed.

In Phase II, current helicopter component structural design practice was reviewed. Visits were made to helicopter manufacturers who have had experience in the design and fabrication of military helicopters to review and record their current failsafe/safe-life design practices and their opinions of present and proposed failsafe/safe-life design criteria. Visits were also made to appropriate Government agencies to obtain their opinions concerning failsafe/safe-life design criteria.

In Phase III, the information collected under Phases I and II were analyzed with respect to the specifications MIL-S-8698, Structural Design Requirements - Helicopters; AR-56, Structural Design Requirements (Helicopters); and FAR-27/29, Airworthiness Standards, Rotorcraft. Deficient areas are identified, and recommendations for corrections are presented. The recommended damage-tolerant design criteria for critical helicopter structural components are presented on pages 39 through 42. Design concepts are presented to show methods of improving the damage-tolerant characteristics of critical helicopter structural components with the approximate increases in cost and weight. The design techniques discussed include redundant structure, controlled fracture structure, and failure-indicating systems. The Hughes OH-6A main rotor blade was selected as the structural component to establish the effects on costs and weights in designing to different damage-tolerant criteria and in using various damage-tolerant design techniques.

PHASE I - STATE-OF-THE-ART DETERMINATION

OBJECTIVE

The objective of this determination phase was to assess and determine current state of the art as to the relevant failsafe/safe-life design methodology for application to helicopter component structure. The primary area of interest was helicopter structure; however, fixed-wing design practices and foreign design practices were also reviewed for maximum applicability to helicopters.

LITERATURE MATRIX

Lists of published documents were obtained from various sources, including the American Helicopter Society (AHS), the American Institute of Aeronautics and Astronautics (AIAA), the Society of Automotive Engineers (SAE), the National Technical Information Service (NTIS), the American Society of Mechanical Engineers (ASME), Battelle Memorial Institute, the Defense Documentation Center (DDC), and the National Aeronautics and Space Administration (NASA). From the documents reviewed that pertained to failsafe/safe-life design, criteria, testing, loads, and inspection, 55 reports were selected as representing the latest failsafe/safe-life technology. These reports are to be compared with existing Government specifications under Phase III. The subjects covered by the reports are presented in Table 1. The reports are listed across the top of the table by their reference numbers, and a dot appears under a number if that report contains information on the subject horizontally in line with the dot. For convenience, the literature cited follows Table 1.

TABLE 1. LITERATURE SOURCE/DATA MATRIX

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2.	MIL-A-8866	AIRPLANE STRENGTH AND RIGIDITY - RELIABILITY REQUIREMENTS, REPEATED LOADS, AND FATIGUE.
3.	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS.
4.	AR-56	STRUCTURAL DESIGN REQUIREMENTS (HELICOPTERS).
5.	SD-24J, Vol II	GENERAL SPECIFICATION FOR DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPONS SYSTEMS - ROTARY WING AIRCRAFT.
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9.	CAM 6, Appendix A	MAIN ROTOR SERVICE LIFE DETERMINATION.
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LITERATURE INFORMATION SURVEY

The following is a brief summary of information obtained from the review of published literature listed in Table 1 pertaining to failsafe/safe-life criteria and technology. The topics discussed include safe-life, failsafe, design concepts, inspection importance, and other design criteria.

SAFE-LIFE CONCEPT

Safe-life is defined as the computed length of time that a structural component subjected to fatigue loads can operate in service with an extremely low probability of catastrophic failure. The analysis to establish the safe-life of a structural component requires data on the fatigue loads or stresses that the component is expected to incur during service and the fatigue strength of the component. Fatigue load spectra are based on the expected usage of the helicopter. The actual structural component is fatigue-tested to determine its fatigue strength. The two principal methods of testing to determine the fatigue strength of a component are constant-amplitude testing at various load levels and spectrum or block testing, in which the expected loads per unit of time are repeated until failure. Constant-amplitude fatigue tests are employed for components with relatively simple loadings and load paths, whereas spectrum loading tests are usually conducted on components that contain complex loadings and multiple or otherwise complex load paths.

S/N CURVE

The results of constant-amplitude fatigue testing are expressed in the form of an S/N curve. The S/N curve is a plot of alternating stress, strain, or load (S) versus cycles to failure (N). The minimum number of test specimens required to establish an S/N curve is four. There are several methods used to determine the shape of the mean S/N curve to be drawn through the test points. The mean S/N curve is reduced by either three standard deviations (3σ) or 20 percent, whichever is greater, for use with the fatigue load spectrum to establish the component's safe-life.

FATIGUE LOAD SPECTRA

The helicopter fatigue load spectrum is based on the helicopter's intended operational usage. The spectrumconsists of a complete list of flight conditions simulating every type of operation likely to be encountered by a particular type of helicopter, with the portion of time spent in each condition or the percentage of occurrence specified. Inflight load or stress measurements are determined for all critical structural components for the critical flight conditions of the spectrum. The measured loads along with the percentage of occurrences are used in conjunction with the test S/N curve to establish the component's safe-life.

SAFE-LIFE ANALYSIS

The method most often used to establish safe-lives is based on Miner's Cumulative Damage Theory. The magnitude of loads and the number of cycles per unit time for a flight condition are obtained from the fatigue load spectrum. The allowable number of cycles that a component can sustain at a given load level is obtained from the test design S/N curve. The number of applied cycles divided by the number of allowable cycles gives the amount of fatigue damage for a flight condition for a unit of time. The safe-life is determined by summing the fatigue damage of all fatigue-damaging flight conditions. There are numerous papers and articles that either praise or condemn the use of Miner's Cumulative Damage equation. It is widely used primarily because of its simplicity and the ease with which it can be used to compute a new safe-life if the helicopter's mission profile is changed.

SAFE-LIFE VALUE

The safe-life of a structural component is an analytically computed number and does not account for the actual load or physical environment experienced by the structural component. Recent trends appear to be away from the computed safe-life concept to designing for failsafe, as defined later in this summary. The computed safe-life that meets the requirements of an extremely low probability of failure in service also establishes the fact that over 99.9 percent of the structural components reaching their mandatory safe-life still retain structural integrity. It is presently required that these components be removed from service (although still structurally sound), thereby increasing the cost of maintenance, logistics, and storage. This is further verified by the many parts that have been fatigue tested after being retired from service which showed little or no loss in fatigue strength due to their service experience.

FAILSAFE DEFINITION

One of the first observations concerning "failsafe" is that the term has a variety of meanings and applications, and no well-accepted definition exists. Originally, the most common definition of failsafe, when applied to structural components, was that a portion of a structural component -- such as a bolt, a longeron, or a portion of skin -- could fail, and the remaining structure would still be capable of sustaining a specified static load without a catastrophic failure. In addition to this important criterion, it was recognized that the remaining structure should have a safe-life sufficient to reach an inspection period that would detect the partial failure before additional structure failed, resulting in a catastrophic failure of the total structure.

FAILSAFE ANALYSIS

Failsafe structure basically involves three phases. First, with a partial failure existing, the remaining structure will not fail completely if subjected to a single severe service load. Second, the life of the remaining structure will be sufficient to reach an inspection period that will detect the partial failure. Third, nondestructive inspection techniques that are capable of detecting partial failures are available and implemented. The basic design philosophy of the first phase is to assume that various parts of the structural components have failed and to ensure that the remaining structure is capable of sustaining the maximum design load. The relatively new fracture mechanics methodology is being used as the technical base for predicting the time or life from a partial failure to a catastrophic failure. Fracture mechanics, in its simplest definition, is a discipline that provides a quantitative relationship among applied stress, crack size, part geometry, crack growth, and failure stress. Through fracture mechanics, new structural components are now being designed to give the maximum crack-growth delay time and/or to include crack-stopping or crack-arresting concepts.

INSPECTION IMPORTANCE

The detection of fatigue cracks before they increase in size so as to cause a catastrophic failure is the ultimate control in ensuring the failsafe characteristics of a flight structure. This has been responsible for some manufacturers' developing structural concepts that include failure-indicating systems. These systems include such concepts as pressurized components where the loss of pressure indicates a possible crack and the use of strain gages or crack-detection wires located in areas of maximum strain.

FAILSAFE VALUE

The automatic mandatory retirement of life-limited structural components could be delayed or omitted with updated failsafe design criteria and methodology. The requirements should include maximum usage of structural redundancy, low crack-propagation rates and/or crack-stopping concepts combined with established inspection periods using the latest nondestructive inspection techniques, and, for some structural components, the inclusion of failure-indicating systems. This would allow a longer usage of helicopter structural components, thereby reducing maintenance and associated costs without jeopardizing the structural integrity of the helicopter.

OTHER STRUCTURAL CRITERIA

Helicopter structure where failsafe/safe-life criteria are of prime importance in the design of structural components includes rotor systems, drive systems, and control systems. Fuselage structure and landing gear structure are essentially determined by other design criteria. Fuselage structure is designed primarily by maximum design load conditions combined with the maximum crash survivability structure. However, areas of the fuselage subjected to sizable vibratory loads from dynamic components should be designed to failsafe criteria. The reserve energy landing condition is the primary design condition for the landing gear and fuselage supporting structure.

PHASE II - REVIEW OF CURRENT HELICOPTER COMPONENT STRUCTURAL DESIGN PRACTICE

OBJECTIVE

A review was conducted of current helicopter component structural design practice. Visits were made to helicopter manufacturers who have had experience in the design and fabrication of military helicopters to review and record their current failsafe/safe-life design practices and their opinions of present and proposed failsafe/safe-life design criteria. Also, visits were made to appropriate Government agencies to obtain their current opinions in regard to failsafe/safe-life design criteria. The review covered helicopters that were designed for different mission profiles and were of widely varying sizes and gross weights, as well as performance. The helicopters used different types of rotor systems, blades, hub retention systems, and types of controls.

AGENCIES AND COMPANIES VISITED

The following are the helicopter manufacturers and Government agencies visited:

Agency/Company

Eustis Directorate, USAAMRDL
Fort Eustis, Virginia

NASA
Langley Research Center
Hampton, Virginia

Kaman Aerospace Corporation
Bloomfield, Connecticut

Sikorsky Aircraft Division
Stratford, Connecticut

Boeing-Vertol Company
Philadelphia, Pennsylvania

Naval Air Systems Command
Arlington, Virginia

Bell Helicopter Company
Fort Worth, Texas

GENERAL

The views and opinions expressed by both the Government agencies and the helicopter manufacturers concerning failsafe/safe-life design criteria were, in general, a reflection of the same information derived from the literature review. However, there are areas of differing opinions among both helicopter manufacturers and Government agencies. A synopsis of the discussions and opinions concerning failsafe/safe-life criteria and design practices follows.

SAFE-LIFE

Most manufacturers believe that the present method of establishing safe-lives has been instrumental in preventing catastrophic failures from occurring in service. However, there are some deficiencies in analytically computed safe-lives. The safe-life does not account for the actual load or physical environment experienced by the structural component. Most failures of structural components in the field have been traced to such factors as improper maintenance or maintenance mistakes, manufacturing errors, material defects, fretting, corrosion, and structural deterioration. Manufacturers and the Government agree that it would be desirable to remove structural components "on condition." The question here is how to ensure that there will not be a catastrophic failure between inspection periods required for "on-condition" removal from service.

FAILSAFE DEFINITION

The consensus was that the term "failsafe" does not have a common definition. In fact, the term could have different definitions applied by the same person because of differences in structural components being designed or analyzed. Some concern was expressed as to the legal definition of "failsafe." It was reported that, in some legal actions, the implication was that failsafe design involved designing a component to fail but safely. There was general agreement that the coinage of a new word or term would be desirable to replace the term "failsafe" if it could have a common meaning to all engineers and the public.

It is recommended that "failsafe/safe-life interface criteria" be replaced and encompassed under "damage-tolerant design criteria." The term "damage-tolerant" is to a large degree self-explanatory.

DESIGN OBJECTIVES

Regardless of the criteria, all companies have basically the same objectives in the design of structural components. Among the desirable characteristics to include in the design of structural components are high safe-life and ample margins of safety for static and cyclic loads with high structural reliability during field operation, while still maintaining low weight and reasonable manufacturing costs. Primary structural components can incur damage from numerous sources:

a. Manufacturing or material defects

b. Field maintenance errors

c. Fretting

d. Corrosion

e. Fatigue-originated crack

f. Deterioration

To optimize new designs and correct structural problems in existing designs, manufacturers are aiming to increase the tolerance to damage of structural components. Methods of designing damage-tolerant structure vary, with each manufacturer's method being largely dependent on his experience with past helicopter structural components. Structural designs that have proven to be reliable are retained, and designs that have created problems in the field are changed. Some of the structural features that one attempts to include in the design of structural components to provide damage tolerance are as follows:

a. High residual strength

b. Low-crack-propagation material

c. Redundant load paths

d. Ease of inspection of critical areas

e. Failure-indicating systems

f. Crack-arresting techniques

g. "Safe" failure modes

The primary objective of designing for structural damage tolerance is to maintain the maximum degree of residual strength and so provide the maximum time for an inspection procedure to find the damage or partial failure prior to the failure's becoming catastrophic.

The above design objectives are also the desired objectives of the customers for the helicopters. The customer's main design objective is that the helicopter performance and structural integrity meet the requirements contracted for at a reasonable cost.

PROBLEMS

Problems arise between the manufacturer and the customer when the customer has specified design criteria requiring that these design objectives be accomplished For numerous reasons, it is not possible at present to design all structural components to be completely damage-tolerant. It is even more difficult to prove or substantiate that a design will meet the requirements of providing high residual strength for all modes of failure and the probability of catastrophic failure is extremely remote between inspection periods.

DAMAGE-TOLERANCE DEFINITION

Any discussion of damage tolerance first requires a definition of the word "damage" that includes type of damage, location, and severity. There are varied and diverse opinions among helicopter manufacturers, other manufacturers, and Government agencies as to what constitutes or defines a mode of failure. Some believe, with service experience to back their beliefs, that it is not possible to define all the modes of failure that can occur in a complex structural component. Others believe that only primary modes of failure should be considered. For some structural components, manufacturers have included in their design a failure-indicating system that monitors one mode of failure. Even the simplest component has more than one mode of failure; therefore, monitoring one mode of failure implies a relatively weak element designed into the structure, with assurances that the remaining structural elements of the component, even damaged, are stronger than the weak element being monitored.

DAMAGE-TOLERANCE REQUIREMENTS

It is generally agreed that, for a structural component to be damage-tolerant, two requirements must be met. First, the component must be able to tolerate some damage without a catastrophic failure, and, second, it must be highly probable that the inspections can detect damage or partial failures and so ensure that the initial damage will not propagate to total failure within established inspection periods.

ANALYST REQUIREMENTS

The analyst of a damage structural component has three factors to determine. First, it must be ascertained whether the residual strength is sufficient to carry the applied loads. Second, the amount of time elapsing before the initial damage increases in magnitude to result in a catastrophic failure of the total component must be determined. Third, inspection periods must be established. For the analyst to perform these tasks, he also requires the following:

a. Magnitude and occurrence of the applied load spectrum

b. Acceptable methodology of determining the residual strength of damaged components

c. Acceptable methodology for determining the time from a detectable failure to a possible catastrophic failure

d. A determination of the frequency of inspections

e. The amount and type of structural testing required

f. The amount and detail of analyses required

DAMAGE-TOLERANT LOAD SPECTRUM

Of first consideration is the flight load spectrum to be applied to the damaged component; this is used to determine (1) if the necessary residual strength exists and (2) the time required for the damage to propagate to catastrophic failure. The consensus is that the earlier concept of "fail-safe," in which the maximum applied load is the maximum design limit load condition, should be retained. There are varying opinions as to the use of the applied flight load spectrum and the percentage of occurrence for determining the time from partial to complete failure. Some prefer to maintain the flight load spectrum (mission profile) as used to establish safe-lives, while others believe that the severity of the load spectrum should be reduced. A reason for the reduction of load magnitude is that inspection periods are at approximately 100-flight-hour intervals, whereas safe-lives are usually of several thousand flight-hours; therefore, the probability of incurring peak maneuvering loads is extremely remote, and those peak loads are structurally provided for by defining the failure time as the time when the residual strength equals the maximum design limit load.

DYNAMIC COMPONENTS

Dynamic components present additional problems with respect to applied loads. The vibratory loads in the main rotor blades are largely dependent on the relationship of the blade's natural frequencies to helicopter forcing frequencies. Damage to the blades affects the blade's natural frequencies as well as its structural strength. All manufacturers have computer programs for establishing the magnitude of both static and vibratory loads on rotor blades. The load programs show good correlation for the prediction of static loads, but all agree that the vibratory load prediction could be improved and that predicting the change in vibratory loads due to blade damage and resulting stiffness change is questionable, at best. In the case of one manufacturer, a blade failure originating from the blade trailing edge caused the dynamic characteristics to change, and the change was felt by the pilot. On landing, the pilot inspected the rotor system and found the partial blade failure. This mode of failure acted as a failure-indicating system, warning the pilot of a partial failure before it became catastrophic failure.

CHANGING LOAD ENVIRONMENT

It has been axiomatic that redundant or multiple-load-path structure is failsafe or damage-tolerant. The majority of helicopter fuselage structures have been accepted as being damage-tolerant or failsafe for all modes of failure. As long as it is not necessary to pressurize helicopters, this is probably true. For fixed-wing aircraft, where pressurization of the fuselage is required, redundant or multiple-load-path structure does not necessarily mean failsafe or damage-tolerant structure. Several years ago, British Comets suffered catastrophic fuselage failures due to rapid crack propagation of the fuselage skin. Recently, another mode of failure caused catastrophic failure of a DC-10: a cargo door opened in flight, causing a large differential pressure on the floor structure. This loading probably caused the floor to collapse and fail the control system, resulting in a catastrophic failure. The point is that, as the operation or loading is increased, structure that has proven to be satisfactory in the past may not be satisfactory for the new loading environment.

ANALYTICAL METHODOLOGY

The analytical methodology proposed for determining residual strengths and time to complete failure of structural components involves the use of fracture mechanics. Fracture mechanics, in its simplest definition, is a discipline that provides a quantitative relationship among applied stress, crack growth, part geometry, and failure stress. The use of fracture mechanics has been limited primarily to the analysis of helicopter components where field-service experience indicates that structural problems exist or in modifications of present production helicopter components. Only for helicopters currently being designed is fracture mechanics used during the initial design phase, and the improvement of structural reliability by the use of fracture mechanics is unknown.

THE DIFFICULT QUESTION

The most difficult question to answer is, "What is required to prove, verify, ensure, or substantiate that a structural component is damage-tolerant?" As previously stated, there are several modes of failure for even the simplest structural component; how many modes of failure should be selected for analysis and test, and by what method? With the hundreds of structural components making up a helicopter, combined with the numerous modes of failure relative to each structural component, it should be apparent that the task of substantiating each possible configuration by analyses and tests is enormous, if not impossible. For example, fracture analysis required to determine the crack growth rate and the time for the crack to reach the critical size of a simple flat plate, under a given steady and cyclic load spectrum, requires extensive analysis that necessitates the use of computer programs.

INSPECTION

Technical papers have been presented by persons from both Government agencies and helicopter manufacturers that propose methods of providing an analytical solution for static residual and fatigue vibratory strengths and establishing inspection periods. It is generally agreed that inspection is the key to damage-tolerant structure, in terms of both the ability to inspect and the ability to operate safely between inspection periods. Reasonable correlation between theory and test has been shown for a specific component and type of damage, but more testing is needed to verify and/or modify proposed theoretical analyses. It would be desirable to review the service experience of various helicopters to assess the types and severity of structural damage occurring in service to aid in establishing probable modes of failure.

GOVERNMENT SPECIFICATIONS

Currently, there exist two helicopter specifications that contain criteria for failsafe/safe-life design. The specifications are Aeronautical Requirements -56, prepared by the Naval Air Systems Command, Department of the Navy, and Federal Aviation Regulations 27/29, prepared by the Department of Transportation, Federal Aviation Administration. The design criteria are similar for both specifications, with two major differences: FAR-27/29 provides the option of establishing a safe-life (service life) or designing to a failsafe criterion that requires the residual strength of a partially failed component to equal the maximum limit load and establishing inspection periods where the probability of catastrophic failure is extremely remote. AR-56 requires fatigue lives established for all components to be 5000 or 6000 hours, depending on helicopter class and, in addition, requires meeting failsafe design criteria. The failsafe design criteria provide for two methods of showing compliance. With the first method, the failure of a single structural element will not cause uncontrollable motions of the helicopter and will not reduce the ultimate factor of safety for flight design conditions from 1.5 to a value less than 1.0. With the second method, all partial failures will become readily detectable by inspection, and the inspection interval is such that the probability of catastrophic failure is extremely remote.

MANUFACTURERS' DIFFICULTY

Manufacturers' difficulty, to date, with the criteria requirements of both AR-56 and FAR-27/29 for new or proposed helicopters has been the requirement to formally substantiate that the inspection periods are such that the probability of catastrophic failure is extremely remote. In the case of AR-56, modifications or deviations are requested. For complying with the failsafe criteria, the first method is used in which no single failure of a structural element will cause a catastrophic failure. In the case of FAR-27/29, the method of establishing safe-lives (service lives) is retained. The manufacturers try to include in the design of the structural components as many damage-tolerant concepts as feasible, but they are not planning to formally substantiate the components as meeting the failsafe criteria.

STRUCTURAL MATERIAL SELECTION

The desirable material properties are slow crack propagation and a high ratio of fracture toughness to yield strength for designing damage-tolerant structural components. Slow crack propagation is desired to permit the maximum time for inspection to detect a crack. The higher the fracture toughness of the material, the greater the residual strength for a given crack or flaw size.

COMPOSITES

Composite materials, composed of glass, boron, Kevlar 49, or graphite fibers in an epoxy matrix, are being researched and developed for application in the design of structural components -- in particular, rotor blades.

Composite materials have numerous properties that make them desirable as a damage-tolerant structural material. Among the desirable structural characteristics are the following:

a. High strength to weight ratio

b. High fatigue strength

c. Slow crack propagation

d. High tolerance to scratches or notches

e. Soft, slow-acting failure modes

f. The ability to vary stiffness

g. Control of contour and twist in fabrication

Currently, there is insufficient field experience to verify that the laboratory results from composites will be realized in production helicopters. Of particular concern is the effect of damage, defects, and long-term environmental exposure on failure modes. This has led some manufacturers to combine metals with composites for the final structural configuration.

METAL PLUS COMPOSITE

A means of achieving load-sharing redundancy for rotor blades without significant weight penalty involves the use of a combination of metal and composites. Tests indicate that the metal will fail prior to the composite. Putting the metal on the outside allows it to be easily inspected for cracks and used as a failure-detection system. The composite material slows the crack-propagation rate of the metal and provides residual strength, thereby increasing the length of the inspection periods.

FAILURE-INDICATING SYSTEMS

Failure-indicating systems have been included by some helicopter manufacturers in the design and construction of certain dynamic structural components, particularly main rotor blades. The failure-indicating systems are designed to monitor the probable mode of failure. Two failsafe systems for metal rotor spar blades are currently in production use: the Blade Inspection Method (BIM), developed by Sikorsky, and the Integral Spar Inspection System (ISIS), developed by Boeing-Vertol. Both systems operate on the same incipient failure warning principle. The spar is sealed and then evacuated (for the ISIS system) or pressurized (for the BIM system). If a crack occurs in a spar, the pressure differential is lost, and an indicator connected to the spar at the root of the blade provides a visual failure indication to the ground crew.

Based on their service experience to date, these manufacturers believe that the added complexity and cost of the failure-indicating systems are justified. The systems answer one of the most difficult questions of damage-tolerant design: that of establishing inspection periods, since they provide continuous inspection of a primary mode of failure.

FAILSAFE/SAFE-LIFE CONCEPTS

The failsafe and safe-life concepts are both "life" problems. The real difference in the two is that, for the failsafe concept, inspection intervals are added and related to the crack-propagation time. More important is the fact that the residual life concept is an active procedure requiring action related to service use, whereas the safe-life crack-initiation approach is a passive method in which it is trusted that nothing will happen prior to replacement of the structural component.

BALLISTIC DAMAGE

Ballistic damage requirements are relatively new, and manufacturers' views differ as to their ultimate effect on failsafe/safe-life design criteria. Ballistically tolerant structure requires the use of redundancy, crack-stopping or delaying techniques, and low-crack-growth materials, all part of the primary requirements for failsafe (damage-tolerant) structure. It is the view of some manufacturers that structural components will require more strength than necessary for the undamaged state, thereby reducing the component's operational stresses and reducing the effort necessary to establish required safe-lives. Data on the size of the projectiles and the resulting ballistic damage are needed to determine whether a structural component would require changes beyond what is necessary to meet present design criteria. Information obtained from helicopter operation in Southeast Asia indicates that some currently designed and manufactured dynamic components can sustain a sizable amount of ballistic damage and still continue to operate for the length of time necessary to reach a safe area.

CRASH SURVIVABILITY

It was generally agreed that increased crash survivability requirements have little or no effect on failsafe/safe-life design criteria. The crash survivability requirements apply primarily to the fuselage and landing gear structure and require structural components capable of absorbing energy. The reserve energy condition is the primary design condition for the landing gear and fuselage supporting structure. Helicopter fuselage structure is considered to be failsafe structure by most manufacturers because of its size and multiple redundancy, and it is designed according to maximum design load conditions combined with crash survivability requirements.

CONCLUSION

It was generally agreed that present-day design procedures and criteria have resulted in structurally safe helicopters. The operating environment for helicopters is becoming increasingly severe, and the fleet sizes of both military and commercial helicopters in operation are growing. To ensure the safety of all helicopters, it is becoming more important -- or even essential -- to place greater emphasis on the "damaged" strength of structural components.

PHASE III - ANALYSIS

EARLY STRUCTURAL DESIGN CRITERIA

Early structural design specifications or regulations dealt primarily with establishing maximum design load criteria. The structural integrity of the aircraft was established by showing that the aircraft structure could sustain the loads resulting from the design load criteria. Methods of analysis were developed to predict the maximum design loads that could occur during the operation of the aircraft. The external loads were distributed by an acceptable methodology to determine the maximum loads on all primary structural elements.

REQUIREMENTS AND SUBSTANTIATION

The essential structural requirements were that there be no yielding at the maximum design (limit) load and that there be no failure at ultimate load, where ultimate load was defined as 1.5 times the limit load. Formal substantiation of compliance with the specifications consisted of structural analyses and static load tests of the completed aircraft. The static tests consisted of testing several of the most critical design load conditions, with the condition determined to be the most critical (from limit load tests) tested to ultimate load. The combination of the structural analyses and successful completion of the static tests substantiated the structural integrity of the aircraft.

ORIGIN OF "FATIGUE" IN THE REGULATIONS

Research and study on the phenomenon of fatigue in metals date back several decades, and, over the years, various theories with mathematical interpretations have been presented on this problem. In the field of civil aviation, references to "fatigue" are found in the earliest publications. Early regulatory material containing reference to the problem includes the following:

1. "The Handbook for Airplane Designers," which was issued in 1927 by the Department of Commerce, stated that care must be taken to avoid the use of standard eyebolts on control systems and surfaces where vibration might cause fatigue failure.

2. In 1934, detailed airworthiness standards were issued by the Department of Commerce (in Aeronautics Bulletin No. 7-A), which stated, in part, that care shall be taken toward preventing fatigue failures by proper material distribution and shape in the detail design of members and fittings.

3. In 1945, airworthiness requirements appeared as Part 04. These requirements were reissued by the Civil Aeronautics Board as: (1) Part 03 (now Part 23) for small aircraft and (2) Part 4b (now Part 25) for transport aircraft. These provided, in part, that the design of the structure shall avoid points of stress concentration where variable stresses above fatigue limits are likely to occur.

4. In 1954, the predecessor of the FAA, the Civil Aeronautics Administration, sponsored a study to develop more specific fatigue criteria for transport aircraft.

FIXED-WING AIRCRAFT

The development of the fixed-wing aircraft has preceded the development of the helicopter in both time and quantity produced. Service experience with the fixed-wing aircraft indicated that practically all structural failures were fatigue-related. The early fatigue requirement was general in nature; e. g. , "the effects of sustained vibration and fatigue loads upon the strength of the material shall be included in selecting the allowable strength values for design." The yield and ultimate strengths of materials are easy to determine by tests, whereas fatigue strengths are difficult to establish and vary greatly as used in the fabrication of structural components.

HELICOPTER SPECIFICATIONS AND REGULATIONS

The specifications and regulations for both military and commercial helicopters were based on fixed-wing aircraft specifications and service experience. The chronological development of regulations and specifications is shown in Table 2. The first regulation, CAR-6, dealt primarily with establishing the requirements for static strength of helicopter structural components, with only a general fatigue requirement.

HELICOPTER SERVICE EXPERIENCE

In the design and development of helicopters, fatigue problems occurred in rotating components, particularly blades, hubs, and rotating controls. Some of these problems resulted in catastrophic failures in a relatively short period of time -- on the order of a hundred hours. This showed a difference between the fatigue structural design of helicopters and that of fixed-wing aircraft. The primary fatigue damage occurring in fixed-wing aircraft results from the ground-air-ground (GAG) condition, which is a high-load, low-cycle fatigue condition. For the rotating components of helicopters, relatively low loads and resulting stresses occurring for millions of cycles produce the major fatigue damage.

TABLE 2. DEVELOPMENT OF STRUCTURAL SPECIFICATIONS FOR ROTORCRAFT

Date	Specification	Requirement
1945	CAR-6/7	Static strength
1950	SR-189	Static strength
	CAM-6, App. A	Service life
	MIL-S-8698; MIL-T-8679	Fatigue/service life
1968	FAR-27/29	Service life/failsafe evaluation
1970	AR-56	Fatigue life/fail-safety

COMMERCIAL SERVICE LIVES

The fatigue requirements of CAR-6 were revised to require the establishment of service lives of life-limited components, with the service life to be determined by fatigue tests or by other methods found acceptable by the administrator. Appendix A of CAM-6 outlined fatigue evaluation procedures acceptable to the Federal Aviation Agency for showing compliance with the fatigue evaluation requirements of CAR-6.250. Appendix A was drafted in the early 1950's and issued in December 1962.

MILITARY HELICOPTER SPECIFICATIONS

The development of military specifications, beginning with SR-189, which led to MIL-T-8679 and MIL-S-8698, followed a path similar to that of the commercial regulations. The existing experience was used to formulate requirements. As specifications and regulations were revised or superseded, fatigue requirements became more specific. By 1955, both military and civil rotorcraft documents required that either a service life or a fatigue life be determined.

FAILSAFE CONCEPT

As more service experience became available, it was apparent that structural components could incur damage from numerous sources. During the middle 1950's, this information led to a new structural design concept that was labeled "failsafe" design. The early concept was that a structural component could sustain a partial failure, and the remaining structure could withstand a required load, with the probability of catastrophic failure being extremely remote. The choice of the term "failsafe" was unfortunate, since it is contradictory in nature and no well-accepted definition exists.

Requirements for failsafe structure in helicopters were included in Federal Aviation Regulations Part 27 (FAR-27), effective in 1968, and in Aeronautical Requirement - 56 (AR-56), prepared by the Naval Air System Command in 1970. The failsafe criteria were expanded to include establishment of inspection periods and the time interval from crack initiation to failure.

ANALYSIS OF SPECIFICATIONS

The requirements relating to failsafe and safe-life design criteria are presented in the following paragraphs as extracted from specification MIL-S-8698, Structural Design Requirements - Helicopters: AR-56, Structural Design Requirements (Helicopters); and FAR-27/29, Airworthiness Standards, Rotorcraft. The information collected under Phases I and II was analyzed with respect to the present specifications. The results of the analysis with respect to each specification follow the criteria extracted from each specification.

Rational structural criteria are developed to correct the deficiencies of the existing specifications. The criteria are titled "Damage-Tolerant Design Criteria" and follow the analyses of the present specifications.

MIL-S-8698 (ASG)

> 3.1.9 Fatigue. - The magnitude of stress reversals shall be minimized, and materials and design details shall be used that minimize the possibility of fatigue failure.

3.2.2.2 Design fatigue loading. - The design fatigue loading shall be in accordance with an approved, fatigue design loading schedule. The helicopter and its components, except those covered by applicable specifications, shall be designed for a minimum fatigue life of 1,000 hours.

Analysis

Specification MIL-S-8698 contains only minimal requirements for fatigue design and a minimum fatigue life of 1000 hours. The specification contains no requirements for failsafe design. The design practices of the helicopter manufacturers exceed the requirements of MIL-S-8698.

AR-56

3.1.9 Fatigue. - The structural design of the helicopter shall be such that repeated loads shall not interfere with mechanical operation of the helicopter, affect adversely its aerodynamic characteristics, require repair, or require replacements of components other than as specifically approved by the procuring activity. This requirement applies to the planned service life of the helicopter from the expected typical repeated load environment resulting from flight operations including maneuvers, buffeting, gusts, pressurization (when applicable), taxiing, landing and from repeated operation of all devices.

3.1.9.1 Design Fatigue Loading. - The design fatigue loading shall be in accordance with an approved fatigue design loading schedule based on realistic mission profiles or in accordance with the profile(s) of Table I. These profiles shall be combined with a rational distribution of significant parameters which affect fatigue life including cg, altitude, gross weight, load factor/bank angle, yaw angle, sinking speed, roll angle, pitch angle, take-off-landing speeds, soil conditions, rotor speeds, rotor-hub moments, control loads, torque variations, vibratory loadings, quasi-static loads, landing gear extension-retraction loads and all other pertinent to describing the fatigue loading spectra that the vehicle will be subjected to. Safe life analyses and tests shall be employed to substantiate the helicopter and all its components for a fatigue life specified in 3.1.9.2.

3.1.9.2 <u>Design Fatigue Life.</u> - The design fatigue life of the helicopter and all its components, unless otherwise specified, shall be as follows:

Class I.	-	6000 hours
Class II.	-	5000 hours

3.1.9.3 <u>Fail-Safety.</u> - The complete airframe and all its components shall be constructed so that failure of a single structural element or control element will neither cause catastrophic failure nor preclude safe continuous flight to a normal destination where repairs/corrections can be made. This requirement is in addition to and not in lieu of those specified in 3.1.9.1 and 3.1.9.2. Redundancy, such as alternate load-paths and systems, and other fail-safe principles are required to achieve this capability. If available or attainable information regarding structural characteristics of major systems or power-plant installations (that are not under the control of the airframe manufacturer) are deemed inadequate, the procuring agency shall be promptly notified. For this fail-safe requirement, the airframe is defined as including all of the structural elements of major systems, their supports, and carry-through structures, and all of the structural connecting and supporting elements of power-plant installations, the failure of which will:

a. Cause uncontrollable motions of the aircraft within the speed limits for its structural design.

b. Reduce the ultimate factor of safety for flight design conditions from 1.5 to a value less than 1.0.

<u>Or alternatively the following may apply:</u>

a. It must be shown that all partial failures will become readily detectable under prescribed and acceptable inspection procedures.

*b. The interval between the time when a partial failure becomes readily detectable and the time when any such failure is expected to reduce the remaining strength of the structure to limit loads must be determined.

c. It must be shown that the interval determined under (b) is long enough with respect to the inspection interval that the probability of catastrophic failure is extremely remote.

System fail-safety analyses shall be performed as specified in ARP-926.

*Proposed change of b.

b. The interval shall be determined between the time when a partial failure becomes readily detectable and when such failure is expected to grow to the point where the remaining strength of the structure has been reduced to support only limit loading condition.

3.5.5 Control System Fatigue Strength. - Strength shall be provided in those components between the pilot's control (or irreversible mechanism) and the control horn inclusive for 30 million cycles of 1.5 times the maximum maneuvering flight load or three times the maximum steady-state flight load, whichever is most critical.

Analysis

General

Specification AR-56 contains the requirements for establishing fatigue (safe) lives and complying with failsafe criteria. The primary fatigue requirement is that, with an approved fatigue design loading schedule, safe-life analyses and tests shall be employed to substantiate the helicopter and all its components for a fatigue life of 5000 or 6000 hours. For fail-safety, the complete airframe and all its components shall be constructed so that failure of a single structural element or control element will neither cause catastrophic failure nor preclude safe continuous flight to a normal destination where repairs/corrections can be made. The fail-safe requirements are in addition to the fatigue safe-life requirements. The failsafe criterion offers two methods of showing compliance. With the first method, the failure of a single structural element will not cause uncontrollable motions of the helicopter and will not reduce the ultimate factor of safety for flight design conditions from 1.5 to a value less than 1.0. With the second method, all partial failures will become readily detectable by prescribed and acceptable inspection procedures, and the inspection interval is such that the probability of a partial failure propagating to cause a catastrophic failure is extremely remote.

Manufacturers' Views

Helicopter manufacturers have had difficulty in meeting the fatigue safe-life and failsafe requirements as stated, and as interpreted from AR-56. This has resulted in requests for modifications of, and/or deviations from, the specification. Some examples of words and phrases that have required additional definition or modification are as follows:

a. 3.1.9 Fatigue. - ... repeated loads shall not ... require repair...

b. 3.1.9.3 Fail-Safety. - The complete airframe and all its components shall be constructed so that failure of a single structural element or control element will neither cause catastrophic failure nor preclude safe continuous flight to a normal destination where repairs/corrections can be made. ...

Under paragraph 3.1.9, it is implied that no failure is allowed since no repair is allowed, whereas paragraph 3.1.9.3 allows the failure of a structural element. Paragraph 3.1.9.3 requires that no failure prevent safe continuous flight to destination, which precludes the use of failsafe structural components whose failure would cause an autorotation landing. An example is the Hughes OH-6A helicopter, where a failure of the main rotor drive shaft would only cause an autorotation landing. However, the opinion was expressed that an autorotation landing of a military helicopter in a combat area is a catastrophic failure, since it could result in loss of the helicopter and crew.

A major problem is the definition of the words "complete" and "all" in relation to substantiation of component fatigue lives, construction of components meeting failsafe requirements, and partial failures. The majority-view specification should be more definitive as to what structural components require formal substantiation for fatigue lives and failsafe criteria and what number and degrees of modes of failure are to be substantiated.

Inspection Versus Damage Size

Specifications, published literature, and manufacturers all agree that inspection is the key to failsafe (damage-tolerant) structure. First, it is desirable to have all critical areas of primary structure as accessible to inspection as possible. The two important damage sizes to be determined are the most probable (smallest) size to be detected by the prescribed inspection procedure and the critical damage size where the residual strength equals the maximum design load. This is required for determining the time interval from probable detection to possible failure. The time interval is necessary for establishing inspection periods. Fracture mechanics methodology is the primary method proposed to establish critical damage sizes and to establish the time interval from detection to growth to critical size. However, there is minimal information on the quantitative (minimal) damage size that will probably be discovered by a prescribed inspection and the required frequency of inspection relative to the established time interval. Service maintenance costs and helicopter downtime can grossly increase if the prescribed inspection procedure is complicated and/or the inspection periods are frequent.

Costs

A structural specification is a statement of requirements with which manufacturers shall prove compliance. As specifications have progressed from requiring formal substantiation of only static strengths to adding the establishment of fatigue (safe) lives and, currently, to requiring failsafe structure, the cost to formally substantiate the structural integrity of helicopter structural components has become practically prohibitive.

Recommendation

It is recommended that the formal substantiation of fatigue (safe) lives be reduced or omitted if compliance is shown for the failsafe (damage-tolerant) criteria. The maximum damage-tolerant concepts are to be incorporated in the design of all helicopter structural components; however, only the critical components for the most probable modes of failure are to be formally substantiated by analysis and tests. The recommended damage-tolerant design criteria are presented on pages 34 through 37.

FAR-27/29

27.571 Fatigue evaluation of flight structure.

a. General. Each portion of the flight structure (including rotors, controls, fuselage, and their related primary attachments) the failure of which could be catastrophic, must be identified and must be evaluated under paragraph (b), (c), (d) or (e) of this section.

The following apply to each fatigue evaluation:

1. The procedure for the evaluation must be approved.

2. The locations of probable failure must be determined.

3. Inflight measurement must be included in determining the following:

   (i) Loads or stresses in all critical conditions throughout the range of limitations in 27. 309, except that maneuvering load factors need not exceed the maximum values expected in operation.

   (ii) The effect of altitude upon these loads or stresses.

4. The loading spectra must be as severe as those expected in operation and must be based on loads or stresses determined under subparagraph (3) of this paragraph.

b. Fatigue tolerance evaluation. It must be shown that the fatigue tolerance of the structure ensures that the probability of catastrophic fatigue failure is extremely remote without establishing replacement times, inspection intervals or other procedures under 27. 1529 (a) (2).

c. Replacement time evaluation. It must be shown that the probability of catastrophic fatigue failure is extremely remote within a replacement time furnished under 27. 1529 (a)(2).

d. Failsafe evaluation. The following apply to failsafe evaluations:

1. It must be shown that all partial failures will become readily detectable under inspection procedures furnished under 27. 1529 (a) (2).

2. The interval between the time when any partial failure becomes readily detectable under subparagraph (1) and the time when any such failure is expected to reduce the remaining strength of the structure to limit or maximum attainable loads (whichever is less) must be determined.

3. It must be shown that the interval determined under subparagraph (2) is long enough, in relation to the inspection intervals and related procedures furnished under 27.1529 (a) (2), to provide a probability of detection great enough to ensure that the probability of catastrophic failure is extremely remote.

e. Combination of replacement time and failsafe evaluations. A component may be evaluated under a combination of paragraphs (c) and (d) of this section. For such component, it must be shown that the probability of catastrophic failure is extremely remote with an approved combination of replacement time, inspection intervals, and related procedures furnished under 27.1529 (a) (2).

27.159 Rotorcraft maintenance manual.

a. Each rotorcraft must be furnished with a Rotorcraft Maintenance Manual containing the following:

1. All information that the applicant considers essential for proper maintenance, including replacement times for major components, if replacement is anticipated. Part numbers (or equivalent) must be furnished for which a replacement time is furnished.

2. The replacement times, inspection intervals, and related procedures approved under 27.571, and the part number (or equivalent) of each component to which they apply. This section of the manual must be identified by the title "Airworthiness Limitations." The information and procedures in this section of the manual -

   (i) Must be consistent with the information in the rest of the manual;

   (ii) Must be practicable; and

   (iii) Must indicate where "equivalent" procedures are to be permitted.

b. The information in the "Airworthiness Limitations" section of the manual must be segregated and clearly distinguished from the rest of the manual.

Analysis

Regulation FAR-27 contains the requirements for establishing fatigue (service) lives or complying with failsafe criteria for the critical helicopter structural components. In the case of FAR-27 versus AR-56, there is a choice of either establishing a service life or complying with the failsafe criteria. Helicopter manufacturers still retain the service-life evaluation for formal substantiation because of the excessive cost and complexity of formally substantiating by analyses and tests that established inspection periods are sufficient to ensure that the probability of catastrophic failure is extremely remote. The comments on AR-56 stated under the paragraphs entitled "Costs, Inspection and Recommendations" also apply to FAR-27.

RECOMMENDED DAMAGE-TOLERANT DESIGN CRITERIA

The following paragraphs state the recommended damage-tolerant design criteria for all primary structural components (including rotor systems, controls, drive systems, fuselage, and their related primary attachments) the failure of which could be catastrophic or result in an autorotation landing. These recommendations were derived from the analysis of the present specifications and regulations (MIL-S-8698, AR-56, and FAR-27) with relation to helicopter manufacturers' current design practices and the information obtained in Phase I. The recommended criteria are given in a format such as might be used to incorporate them into MIL-S-8698.

The rationale for each section of the recommended damage-tolerant criteria (numbered to correlate with the criteria themselves) follows the presentation of the criteria. Recommended design criteria for fuselage and landing gear are presented on page 43.

DAMAGE-TOLERANT DESIGN CRITERIA

3.1.1 Fatigue evaluation. - Each portion of the flight structure (including rotors, controls, fuselage, drive systems, and their related primary attachments) the failure of which could be catastrophic, or result in an autorotation landing, must be identified. The primary locations of probable failures are to be determined.

3.1.2 Design fatigue loading. - The design fatigue loading shall be in accordance with an approved fatigue design spectrum based on realistic mission profiles. The spectrum shall include all critical fatigue load conditions including, but not limited to, the following:

a. The critical design gross weights.

b. The main rotor RPM ranges, power on and power off.

c. The maximum forward speeds for each main rotor RPM within the ranges determined under (b).

d. The maximum rearward and sideward flight speeds.

e. The center of gravity limits corresponding to the limitations determined under (b), (c), and (d).

f. The positive and negative limit maneuvering load factors.

g. The effect of altitude.

h. The maximum maneuvering conditions including yawing, rolling pullouts, and turns.

3.1.3 Inflight load measurement. - Inflight loads or stresses are to be determined for each portion of the flight structure identified in paragraph 3.1.1.

a. Loads or stresses in all critical conditions throughout the spectrum of paragraph 3.1.2, except that maneuvering load factors need not exceed the maximum values expected in operation.

3.1.4 Fatigue life. - The fatigue life (safe-life) is the period of time that a structural component may operate subjected to the fatigue loading of paragraph 3.1.2, where the probability of catastrophic failure is extremely remote. Formally substantiation of fatigue lives (safe-lives) is not required if compliance is shown for the damage-tolerant criteria under paragraphs 3.1.5 thru 3.1.10, unless the fatigue life is to be used in conjunction with the establishment of inspection periods under paragraph 3.1.6 (c).

3.1.5 Damage-tolerance definition. - Damage-tolerant structure is defined as any primary structure whose characteristics are such that in the presence of abnormalities, such as fatigue cracking, physical damage, deterioration, fabrication errors, or material flaws, the structure will meet the following requirements:

a. The probability of a catastrophic failure prior to detection of the abnormality is extremely remote.

b. The structure will continue to function satisfactorily and shall not flutter, vibrate uncontrollably, or diverge unstably for any loading condition within the normal operating placard limits for the helicopter.

3.1.6 Damage-tolerance evaluation. - The following apply to damage-tolerant evaluation:

a. It must be shown that all partial failures will become readily detectable under prescribed and acceptable inspection procedures.

b. The interval between the time when any partial failure becomes readily detectable under subparagraph (a) and the time when such failure is expected to grow to the point where the remaining (residual) strength of the structure has been reduced to the maximum design limit load must be determined.

c. It must be shown that the time interval determined under subparagraph (b) is long enough with respect to the inspection periods that the probability of catastrophic failure is extremely remote.

3.1.7 Damage-tolerant loads. - The minimum load requirement used to determine the time interval under pagraph 3.1.6 (b) is the fatigue load spectrum established under paragraph 3.1.2 less 10 percent of the maximum loads of the spectrum.

3.1.8 Degree-of-damage requirements. - The following are the amounts of damage or failure to be used for damage-tolerant evaluation under paragraph 3.1.6:

a. The damage sizes to be considered probable of discovery, unless an inspection technique is demonstrated to discover smaller size damages, are as follows:

   1. A crack length of 0.05 inch at the edge of a structural element.

   2. A crack length of 0.25 inch located in a structural element away from the edge.

b. The minimum amount of damage or failure that a structural component shall sustain and still retain the structural strength to react maximum design load is for:

   1. Multiple-element structure - The complete failure of any one structural element.

   2. Monolithic or quasi-monolithic structure - A visible crack length of 1 inch or 10 percent of the component's width, whichever is greater.

3.1.9 Damage-tolerance substantiation. - Meeting the damage-tolerance requirements under paragraphs 3.1.5 and 3.1.6 is to be formally substantiated for only the most probable modes of failure. Fracture analysis methodology, combined with tests of full-size components or small-scale test specimens that simulate the final component, is an acceptable procedure to formally substantiate the damage-tolerance requirements of residual strength and time intervals under paragraph 3.1.6. Other equivalent methods of substantiation may be used where their applicability can be justified.

3.1.10 Inspection periods. - The maximum inspection period cannot exceed one-fourth of the established time interval under paragraph 3.1.6 (b).

RECOMMENDED DAMAGE-TOLERANT DESIGN CRITERIA RATIONALE

3.1.1 Fatigue. - The requirement is to identify the critical structural components of the helicopter to which the damage-tolerant criteria apply. The locations of probable failures are to be determined for each critical component to aid in establishing inspection procedures required to monitor the areas of probable failure.

3.1.2 Fatigue loading. - Fatigue loading establishes the fatigue design load spectrum required to establish fatigue lives (safe-lives), when necessary, and to be used in determining the time interval from detectable partial failure to the time when the residual strength is reduced to a value equalling the maximum design load.

Currently, studies are in progress to define realistic mission profiles of various military helicopters. The studies are to recommend flight conditions and percentages of occurrences, and, if the results of the studies are accepted, the results could be included or referred to in this paragraph. Mission profiles are being established for the following types of helicopters: observation, utility, utility/tactical assault, attack, crane, and transport.

3.1.3 Load measurement. - Inflight load or stress measurements are necessary to verify the analytically predicted loads used in the initial design of the helicopter structural components, to establish fatigue lives, and/or to show compliance with the damage-tolerant criteria. The load or stress measurements should be obtained at the critical areas of each structural component, whenever possible, or in a manner in which the extrapolation of the load to the critical area is highly reliable.

3.1.4 Life. - The fatigue life (safe-life) is an analytically computed period of time during which the probability of catastrophic failure of a structural component is extremely remote. The safe-life approach is a passive method which trusts that nothing will happen prior to replacement of the structural component. The recommendation is to maintain reasonable fatigue lives but to reduce or eliminate the requirement for formal substantiation by analyses and tests.

3.1.5 Definition. - This section defines the primary requirements that structural components shall meet to be damage-tolerant.

3.1.6 Evaluation. - This section defines the three factors that must be determined in the evaluation of damage-tolerant structures. The damage-tolerant design criteria's important relationships are illustrated schematically in Figure 1. The percentage of design load and the damage size are plotted on the vertical axis. Cyclic or time-dependent criteria are defined as time along the horizontal axis. The allowable strength will decrease as the damage size increases, resulting in the solid curve, concave downward. The cumulative load history versus time is the dashed curve, concave upward. Damage size as a function of time or cumulative cyclic loading is the solid curve, concave upward.

The key damage sizes are indicated on Figure 1 as points 1, 2, and 3. The time to initiate and grow a fatigue crack to inspectable size is shown as point 1; growth to the most probable size of discovery requires additional time to point 2. The critical damage size is point 3, where the residual strength is equal to the maximum design load. The inspection interval is the time required for the crack to grow from point 2 to point 3 and is the time available for inspection and repair or replacement. This time interval is also used for establishing the inspection periods.

3.1.7 Loads. - This section defines the minimum load requirements to be used in the damage-tolerance evaluation under paragraph 3.1.6. The reasoning for the reduction of load magnitude is that inspection periods are approximately 100-flight-hour intervals, whereas safe-lives are usually of several thousand flight-hours. Therefore, the probability of incurring peak maneuvering loads is extremely remote, and the peak loads are structurally provided for by defining the failure time as the time when the residual strength equals the maximum design load, a load occurrence expected only once in the lifetime of the helicopter.

3.1.8 Damage size. - There exists only very minimal information concerning the sizes of damages or cracks to be considered probable of discovery. The key to damage-tolerant structure is the size of damage considered to be probable of discovery, which is of prime importance in establishing the time interval to possible failure and the inspection frequency. It is recommended that a study be conducted of the latest state of the art of field-service inspection, with emphasis on

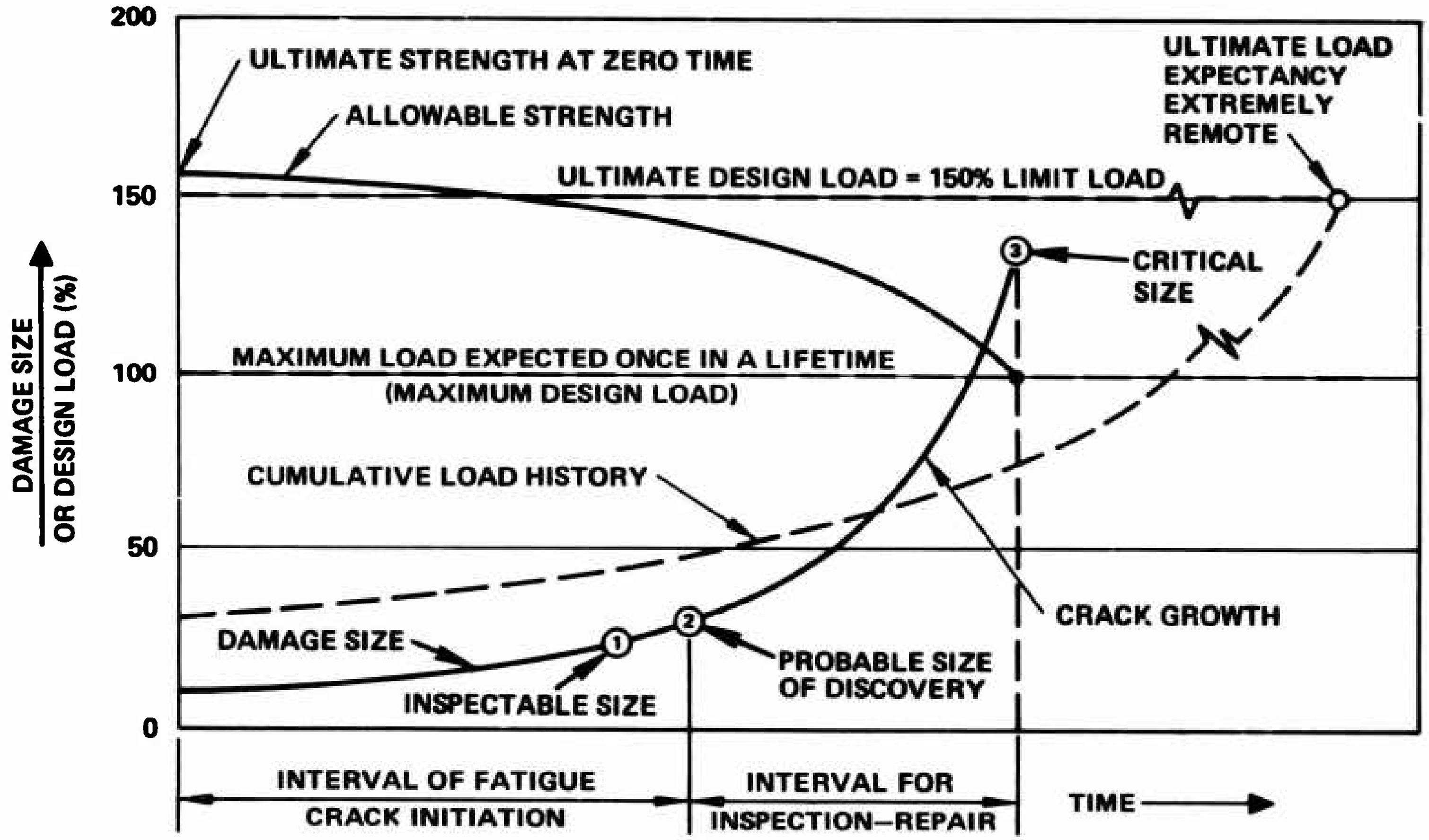


Figure 1. Relationships of Applied Load History, Strength, and Damage Size to Time.

the types and sizes of damages that are probable of discovery. A comparison of various inspection methods listing the advantages and disadvantages of each method is presented in Table 3 from Reference 12.

3.1.9 Substantiation. - The most costly item in designing to damage-tolerant (failsafe) criteria is the showing of compliance or formal substantiation that the helicopter structural components meet the criteria requirements. The costs of substantiating fatigue lives (safe-lives) is increasing to the point of becoming prohibitive. The high cost of the safe-life concept lies in the full-scale testing requirements, which not only consume costly components but also require expensive facilities and long periods of testing. One helicopter manufacturer stated that the components involved in rotor head and shaft tests are valued at 11 percent of the helicopter fly-away cost, and at least four undamaged specimens are required to develop the strength of one configuration (Reference 25). As previously stated in this report, there are several modes of failure for even the simplest structural component. It should be apparent that to establish by tests the time interval from initial damage to failure -- the important element of damage-tolerant design -- for all modes of failure for all critical structural components is economically infeasible and probably impossible.

The results of this study clearly indicate that, unless an economically feasible method of substantiating damage-tolerant structure is approved by both Government agencies and helicopter manufacturers, the manufacturers will continue to request deviations from specifications and will design to the concepts of a computed safe-life and the failsafe criterion that does not involve establishing inspection intervals. This is true even though both customer and manufacturer agree that the most important structural characteristic of a component is its "damaged" strength.

3.1.10 Inspection periods. - After the time interval from detectable damage to reduction of strength to maximum design load is established, the remaining question concerns the frequency of inspection. The recommendation is that the inspection period not exceed one-fourth of the established time interval. This would allow a minimum of three inspections prior to the residual strength of the component being reduced to equal the component's maximum design load, the load expected to be experienced only once in the helicopter's lifetime.

TABLE 3. COMPARISONS OF VARIOUS INSPECTION METHODS

Method	Advantages	Disadvantages
Visual	1. Cheapness. 2. Portability. 3. Immediate results. 4. Minimum special skill. 5. Minimum part preparation.	1. Suitable only for surfaces which can be viewed. 2. Generally detects only larger defects. 3. Misinterpretation of cracks and scratches.
Radio-graphic X ray	1. Ability to inspect for both internal and surface defects. 2. Ability to inspect parts covered or hidden by other parts or structure. 3. Permanent test record obtained. 4. Minimum part preparation required.	1. Most expensive. 2. Airplane may have to be defueled. 3. Area must be cleared of other personnel to avoid X-ray exposure. 4. Test method is highly directional; depends on crack/X-ray source orientation. 5. High degree of skill required for varied technique development and radiographic interpretation.
Radio-graphic isotopes	1. Portability. 2. Needs less area to gain access for energy source. 3. Can accommodate thicker material sources. 4. Less expensive than X ray.	1. Must conform to Atomic Energy Commission regulations for handling and use.
Eddy current	1. Portability. 2. Moderate cost. 3. Immediate results. 4. Sensitive to small indications. 5. Little part preparation.	1. Essentially a surface inspection. 2. Surface to be inspected must be accessible to contact by the eddy current probe. 3. Rough surfaces interfere with test sensitivity. 4. Suitable for inspection of metals only. 5. No permanent test record. 6. Considerable skill and familiarity required in handling test equipment. 7. Time-consuming to scan large areas.
Ultra-sonic	1. Suitable for surface and subsurface defects. 2. Sensitive to small defects. 3. Immediate test results. 4. Little part preparation. 5. Wide range of material thicknesses can be inspected.	1. Surface of part to be inspected must be accessible to sonic probe. 2. Rough surfaces interfere with test results. 3. No permanent test record. 4. Test method is directional depending on sound beam-defect orientation. 5. High degree of skill and experience required to set up and interpret results for varied test conditions.
Dye penetrant	1. Cheapness. 2. Portability. 3. High sensitivity. 4. Immediate results. 5. Minimum skill required to perform.	1. Can only inspect surface of parts accessible to penetrant application. 2. Defects must be open to surface. 3. Part preparation, such as removal of finishes and sealant, required. 4. No permanent test results. 5. Direct visual detection of results required. 6. Requires a high degree of cleanliness for satisfactory inspection.
Magnetic particle	1. Semiportable. 2. Sensitive to small indications. 3. Detects surface and near-surface defects. 4. Sensitive to inclusions as well as cracks. 5. Moderate skill required to perform.	1. Only suitable for ferromagnetic material. 2. Part must be physically and visually accessible to perform test. 3. Removal of most surface coatings and sealant required. 4. Inspection is semidirectional, requiring a general orientation of field to defect. 5. No permanent test results unless the indications from dry powder technique are recorded by pressing "scotch" tape on the surface. 6. Not usable in areas where a strong magnetic field may damage instruments. 7. Part must be demagnetized after inspection.

FUSELAGE DESIGN CRITERIA

Helicopter fuselage structure is generally accepted by both Government agencies and manufacturers as being damage-tolerant (failsafe) structure because of the fuselage size and multiple redundancy. The areas of the fuselage structure reviewed for possible fatigue load damage are the areas where dynamic components attach to the fuselage. The dynamic components include the rotor systems, control systems, engine and drive systems, and other components that produce sizable vibratory loads to be reacted by the fuselage. For these areas, the structural design criteria should be the damage-tolerant criteria presented on pages 43 through 47.

The design criteria for helicopter fuselage structure remain essentially the same, consisting of designing to maximum design flight and landing load conditions combined with increasingly severe crash survivability requirements. However, should it become necessary to pressurize helicopter fuselages, present fuselage structures would probably not be structurally acceptable, requiring that additional damage-tolerant features be included in the design. This additional requirement has already been experienced by the manufacturers of fixed-wing aircraft.

LANDING GEAR DESIGN CRITERIA

The landing gear is designed primarily by static load conditions involving various landing configurations, taxiing conditions, and ground handling conditions. The landing gear is also designed to provide the maximum ability to absorb energy during hard or crash landings, thereby adding to the crash survivability of the helicopter.

DAMAGE-TOLERANT DESIGN CONCEPTS

INTRODUCTION

Damage-tolerant design concepts are presented for various helicopter structural components in this section of the report. The concepts include redundant structure, controlled fracture structure, and failure-indicating systems. The helicopter components discussed include the following:

a. Rotor blades

b. Hub and blade retention systems

c. Rotating primary flight controls

d. Propulsion system

In the presentation of the damage-tolerant concepts, the approximate penalties in cost and weight are discussed. It is beyond the scope of this study to perform a detailed cost-weight trade-off study and life-cycle cost analysis for all the design concepts presented. The Hughes OH-6A main rotor blade was selected as the structural component to establish the effects on costs and weights in designing to different structural criteria. The study also includes the effects of using different damage-tolerant design techniques. The cost and weight study starts on page 72.

The design techniques used by aircraft manufacturing companies to design structure and components to include damage-tolerant concepts are based on their own engineering and manufacturing experiences and know-how, and therefore they often vary markedly in design approaches. If this were not the case, helicopters for certain functions and of various sizes and weight classifications would be much more alike in detail design than at present.

ROTOR BLADES

METAL ROTOR BLADE

The design, development, and testing of rotor blade structures are among the most difficult and involved procedures connected with producing helicopters. The aerodynamic environment causes severe interacting fatigue loads along with the high steady loads in most of the major parts of a blade assembly. Any minute cracks or crack-producing damage to the main structural elements of a blade can progress to a point of failure over a short period of time. The determination of possible methods for taking care of these types of failure contingencies in practical and cost-effective approaches is the main objective of this part of the failsafe/safe-life study.

One approach for study involves taking contempory blade designs and investigating methods of obtaining increased damage tolerance. There are numerous structural arrangements and configurations that can be used for various advantages and degrees of damage tolerance. The type of structure selected for a blade will be influenced to a great extent by the previous experience and manufacturing capabilities of the company making the blade. Rotor blades for the newer military-type helicopters that are likely to operate in a combat environment are being designed with vulnerability/survivability features incorporated. These will obviously provide many damage-tolerant characteristics.

The effect of helicopter size is an influential parameter when considering designs for damage tolerance. In general, it is easier to supply redundancy on the larger size helicopters; the main components can be more economically assembled with a multiplicity of pieces that provide various degrees of redundancy for these components. On smaller helicopters, the cost of producing and assembling a greater number of pieces will make the effort for redundancy more expensive.

Figure 2 is a cross section of a typical production-type D-spar blade design. Here, the single-piece spar carries the major combination of loads, which classifies it as a quasi-monolithic structure. This spar type can be an aluminum alloy extrusion where the wall thickness may be varied as

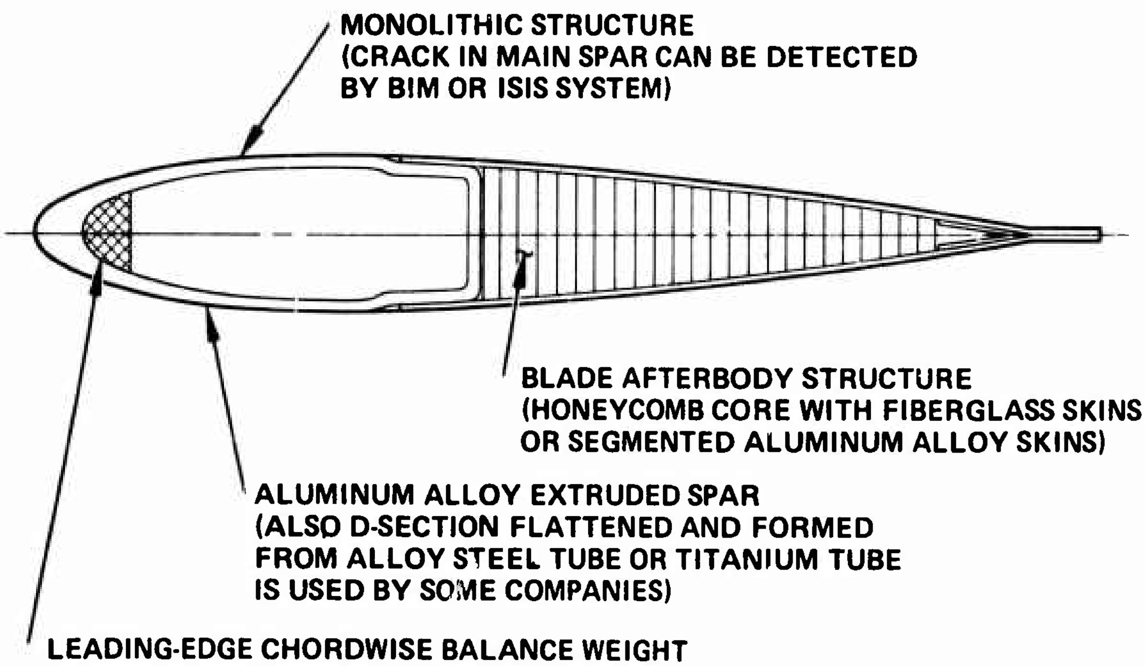


Figure 2. D-Spar Type Contemporary Blade.

needed and joggled lands provided as shown to attach the blade afterbody fairing. Additional shear webs can be provided, creating a multiple-cell spar extrusion where necessary on larger blades.

Another method of producing a very similar spar is to swage, roll, and flatten a special machined steel or titanium tube. The spar wall thickness will be thinner due to the material densities and strength properties, and this method permits the root end of the tube to be retained in a circular shape for easier attachment of the root end fitting or fittings. The tube can gradually taper while changing from a circular section to a leading-edge airfoil section, which reduces the stress discontinuity and concentration effects.

A blade with monolithic structure of this general design is one of the simplest and least expensive to manufacture in production quantities. Obviously, this is one of the prime reasons for its wide use at the present time. Increased interest in more damage-tolerant-type structures is motivating companies to look into new blade structural arrangements for their new helicopter projects.

Figures 3 through 6 illustrate some ideas and methods of providing redundancy and crack-arresting structure to rotor blades. There are many possible combinations of different parts added and arranged to achieve various degrees of redundancy. These sketches give only a limited number of suggestions for approaches to the failsafe problem.

General conclusions onthe cost and weight aspects of the designs shown will follow after a brief discussion.

Figure 3 is a sketch of a blade with a dual spar arrangement. These are simple box-section spars spliced together with a single or double C-section outer skin arrangement. The basic objective here is to fabricate the spar assembly from several pieces and, thereby, obtain redundancy. The parts should be sized so that an adequate fatigue life would remain after failure of one part to provide a sufficient length of time for detection of the failed part.

There are some obvious advantages in redundancy when using several layers of material for the C-spar skin. However, material, material processing, and fabrication costs will be significantly higher than for a single skin.

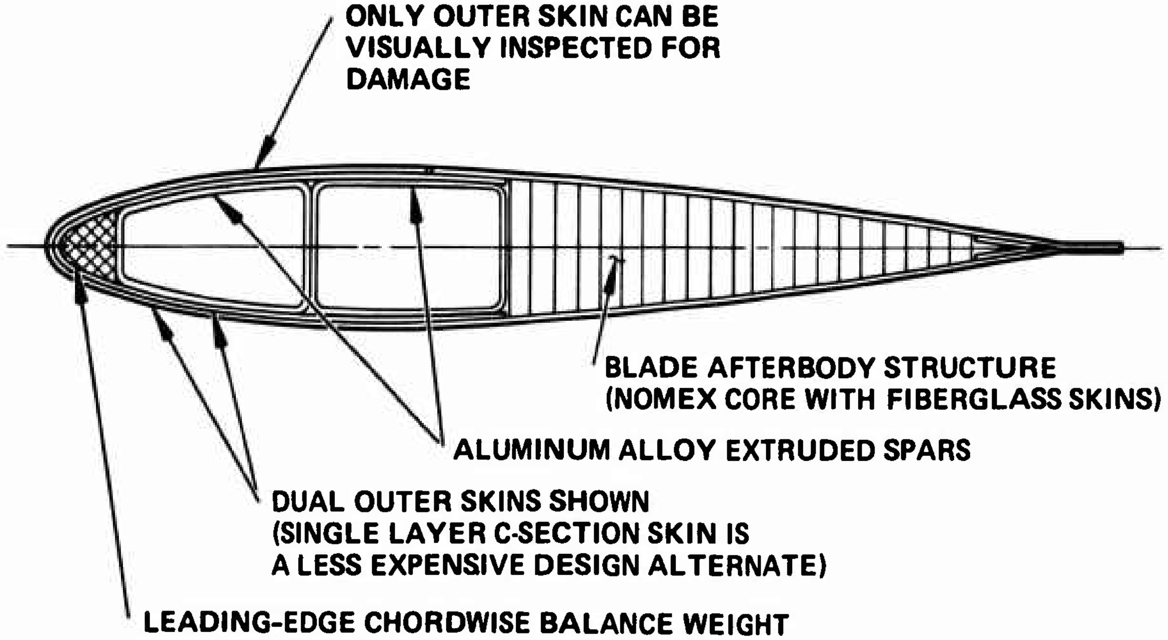


Figure 3. Dual Extruded Spar Design.

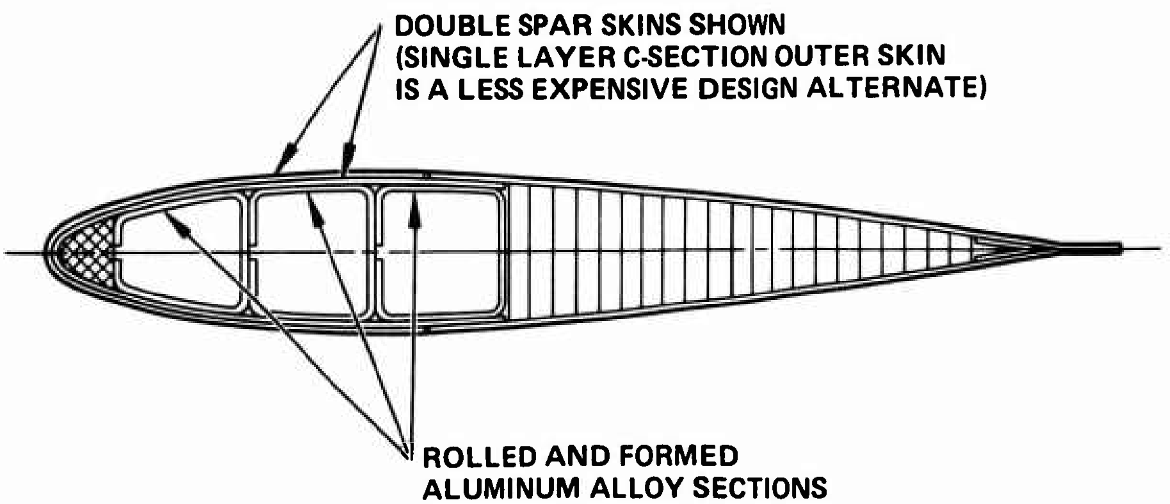


Figure 4. Triple Rolled Section Spar Design.

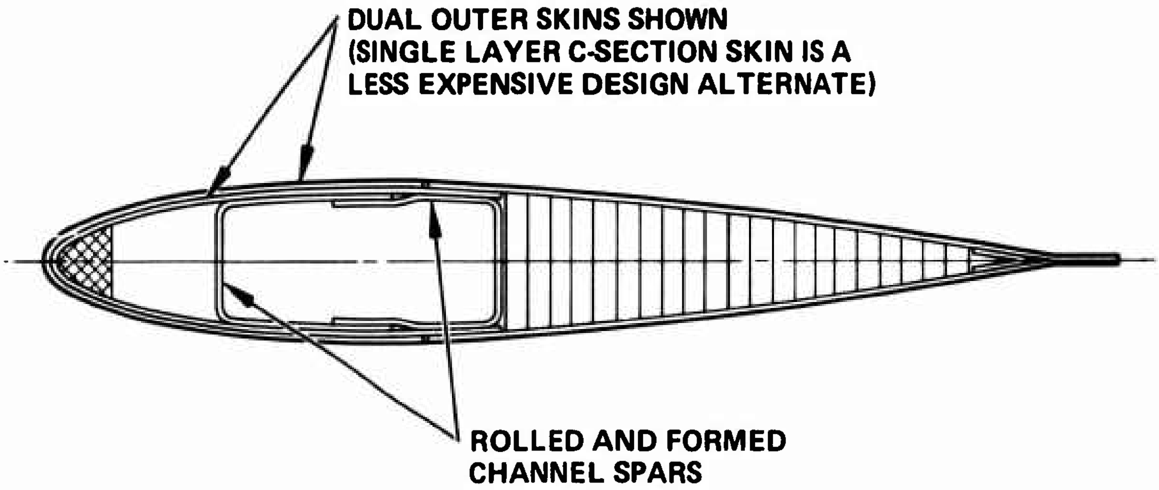


Figure 5. Rolled C-Section Spar and Skin Design.

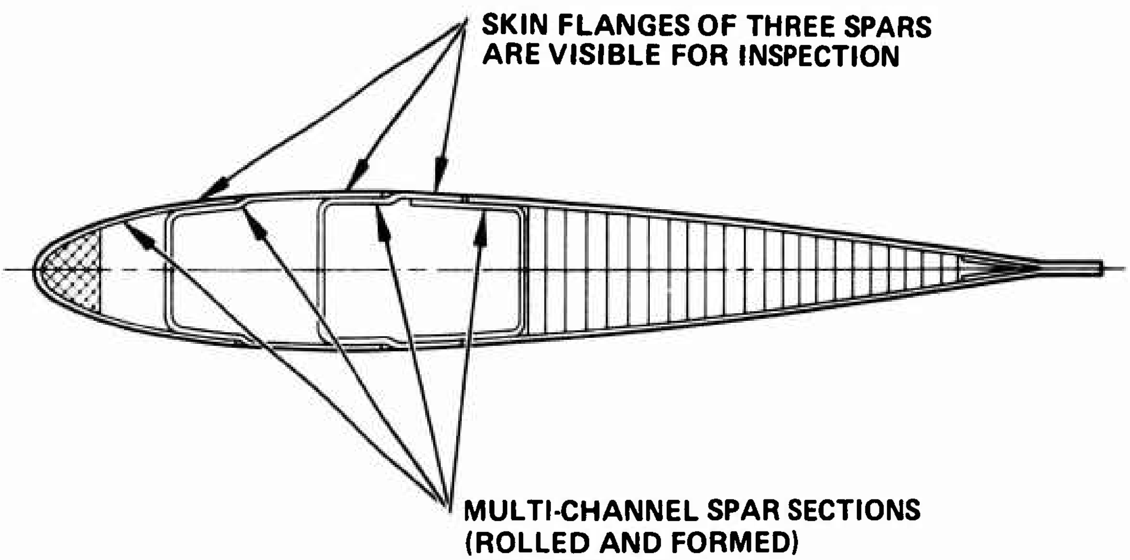


Figure 6. Multi-Channel Spar/Skin Design.

One disadvantage of this and similar structural arrangements is that only the outer skin layers can be visually inspected for fatigue cracks. Dents, scratches, and other forms of externally caused damage can obviously be found by inspection of the outer surfaces. A failure of an internal member has to be found by some type of failure-indicating system or recognized by the crew detecting a change in blade stiffness due to its going out of track or causing other abnormal vibrations. The outer skin along the maximum airfoil thickness region is the highest fatigue-loaded part of the basic blade structure. On a properly designed and developed blade, the first fatigue crack should start developing at an aft edge of this outer skin where it can be found by visual inspection.

Figure 4 is a blade design similar to Figure 3, except that a three-cell arrangement is used instead of two. The spars proposed are rolled and formed sections instead of extrusions. This scheme would permit somewhat thinner webs, and, therefore, the third spar could be used without excessive weight penalties. This would add to the number of redundant members and should result in an increase in the remaining fatigue life after a member is damaged.

This arrangement would make the blade more expensive to build than the one previously discussed (Figure 3). The added member would increase the internal tooling needed during the blade bonding operations. Also, the rolled and formed sections shown would be more expensive to manufacture to the close tolerances needed for rotor blade use.

Figure 5 is a structural configuration assembled from rolled channel sections. Here, redundancy is provided in addition to crack-arresting, overlapping joints. The design can be accomplished with simpler internal tooling during the main bonding operation than with the previous configuration. In addition, open C-section channels are easier to form than the closed type.

Figure 6 is another structural configuration assembled from rolled and formed channel sections. Here, four basic channels, including the nose section, make up the complete spar structure. One advantage of this particular arrangement is that the crack-susceptible edges of the three forward channel sections are on the blade outer surface, where they can be visually inspected. Only the flanges of the aft channel that close the spar assembly box section are not visible for inspection.

Forming and assembly tooling required must be more precise for this type of blade construction in order to maintain the airfoil contour within acceptable production tolerances.

Comparing these four damage-tolerant-type designs costwise and weightwise against an efficiently designed and developed quasi-monolithic blade configuration with limited life (similar to Figure 2), the following conclusions can be drawn:

a. Except in the very large blade-size range, the cost of building rotor blades is higher with the addition of redundant members and/or the addition of crack-arresting members or material. This is primarily due to the increase in the number of parts that must be made, processed, inspected, and assembled, as well as to the increase in tooling complexity required.

b. If actual redundant members or crack-arresting members or materials are added to a rotor blade structure principally to provide damage tolerance, the blade weight will be increased. This weight increase will be directly related to the amount and method of redundancy provided.

METAL AND COMPOSITE BLADE

The metal and composite blade structure consists of a steel skin and channels bonded to each other and to the composite tubes. A cross section of the blade is shown in Figure 7. The skins and channels would be manufactured from thin, high-strength stainless steel. The composite tubes would be manufactured from glass, Kevlar 49, or graphite fibers in an epoxy matrix. The composite tubes would be essentially unloaded

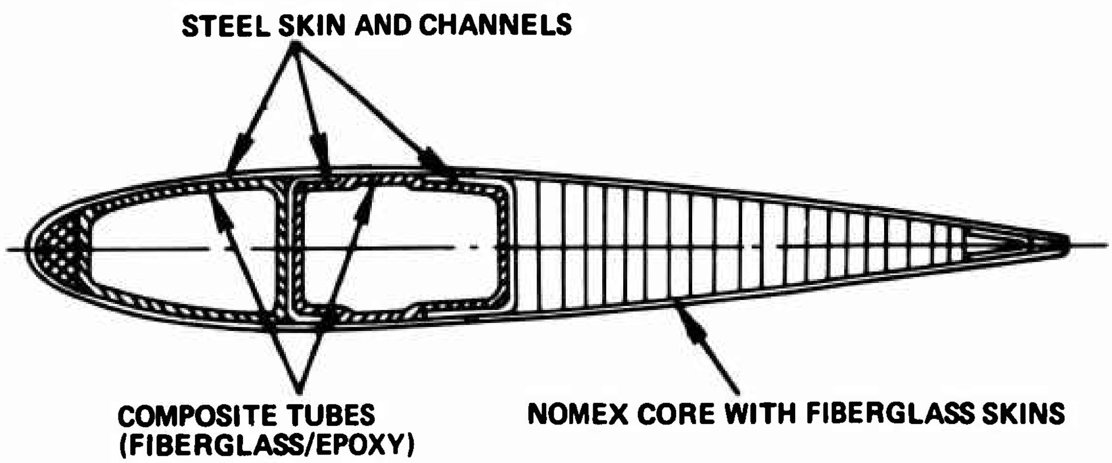


Figure 7. Metal and Composite Blade Design.

during normal operation of the undamaged blade because of the lower modulus of elasticity of the tubes. However, should the steel skin or channels sustain damage, the composite tubes would both provide load redundancy and act as a crack arrestor. These features should provide ample time for even visual inspections to discover partial failures prior to the failures' propagating to cause a catastrophic failure.

The metal and composite blade concept has desirable features in the event of ballistic damage or tree strikes. Metals tend to tear when impacted by projectiles, and the effect of the composite tubes would be to limit this tearing action. In the event of tree strikes, the steel skin supported by the composite tubes would sustain less damage than either blades constructed of all steel or all composites. This blade concept is used in the cost and weight trade-off study of the OH-6A blade; for its relative life-cycle cost, refer to pages 75 and following.

COMPOSITE ROTOR BLADE

Helicopter rotor blades of composite materials (primarily glass/epoxy) have been in limited production for several years. Reports on the performance and service life of fiberglass blades have been very encouraging. Some helicopter manufacturing companies feel that composite blades will soon become mandatory to achieve the high levels of safety, survivaability, and low operating costs that will be required on new helicopter projects.

The main attributes of composite materials that make them so attractive to rotor blade designers are the following:

a. High static tensile strength-to-weight ratios.

b. High allowable fatigue strengths.

c. High notched specimen fatigue strength.

d. Soft, slow-acting failure modes.

In addition, designing rotor blades using filamentary composites gives the designer additional means for obtaining and adjusting the desired strength, stiffness, and dynamic properties. Physical parameters can be tailored by the placement and orientation of the load-carrying fibers. This ability to build directional properties into the blade can result in greatly improved structural efficiency when properly utilized. Another beneficial feature with composite sandwich construction is the ability of a blade to be manufactured and to retain its airfoil aerodynamic profile under flight loads.

Helicopter main rotor blades must retain adequate rotational inertia for satisfactory autorotational landing characteristics in case of power failure. These lightweight, high-strength materials permit extra load-carrying fibers to be added to the blade, which greatly increases its damage tolerance and vulnerability/survivability capabilities when compared to an equivalent-weight metal blade.

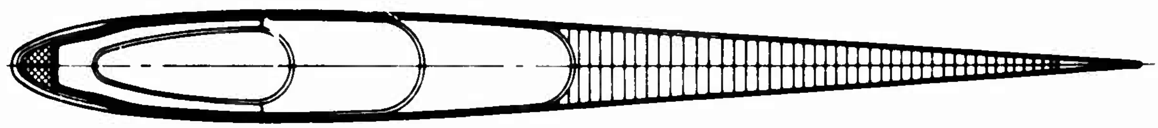
Figure 8 illustrates some composite blade designs that Hughes Helicopters has been studying for possible use on an advanced blade program. The primary structural elements are arranged to achieve low vulnerability and high survivability from gunfire; therefore, they should be highly tolerant to all other forms of damage.

Figure 9 is a more detailed isometric view of the design concept shown in Figure 8C. The composite spar tubes form a multi-cell, box-beam inner structure. The longitudinal unidirectional filaments that carry the major centrifugal forces and the flapwise bending loads are supported between the top and bottom surfaces of the spar tubes and the outer skin. This arrangement permits the longitudinal filaments to be spread chordwise a maximum amount in order to reduce the vulnerability to gunfire damage.

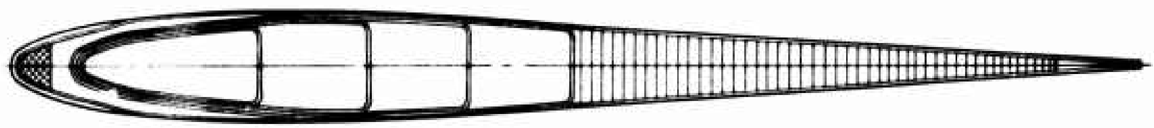
The main continuous longitudinal (spanwise) filaments can be wrapped around a bushing at the root end, as shown in Figure 10, for forming a blade attachment fitting with a single main pin. Other forms of fittings can also be used with the wraparound concept. The main structure is thus highly redundant, with minimum load-path discontinuities. Smaller chord designs (especially tail rotor blades) can utilize simpler structural arrangements in the basic blade cross sections.

The trailing-edge unidirectional filaments can also be wrapped around a bushing and the root end rib built up in this area as illustrated for a drag brace attachment point.

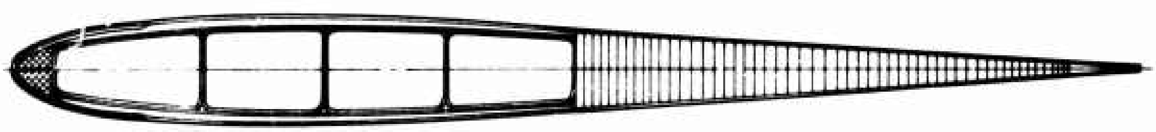
In summary, composite materials make it possible to design more damage-tolerance features into rotor blades. Also, the multi-fiber/bonding matrix makeup of the basic materials is inherently much less notch-sensitive.



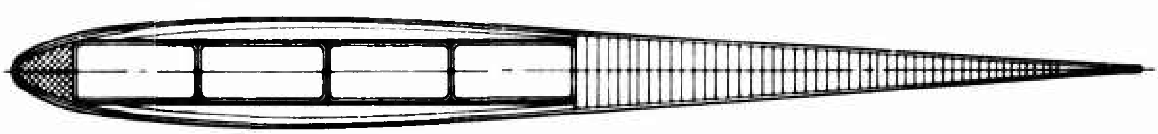
A. STEPPED THREE-TUBE SPAR DESIGN



B. STEPPED FOUR-TUBE SPAR DESIGN



C. TRAPEZOID FOUR-TUBE SPAR DESIGN



D. RECTANGULAR FOUR-TUBE SPAR DESIGN

Figure 8. Damage-Tolerant Composite Blade Structure Concepts.

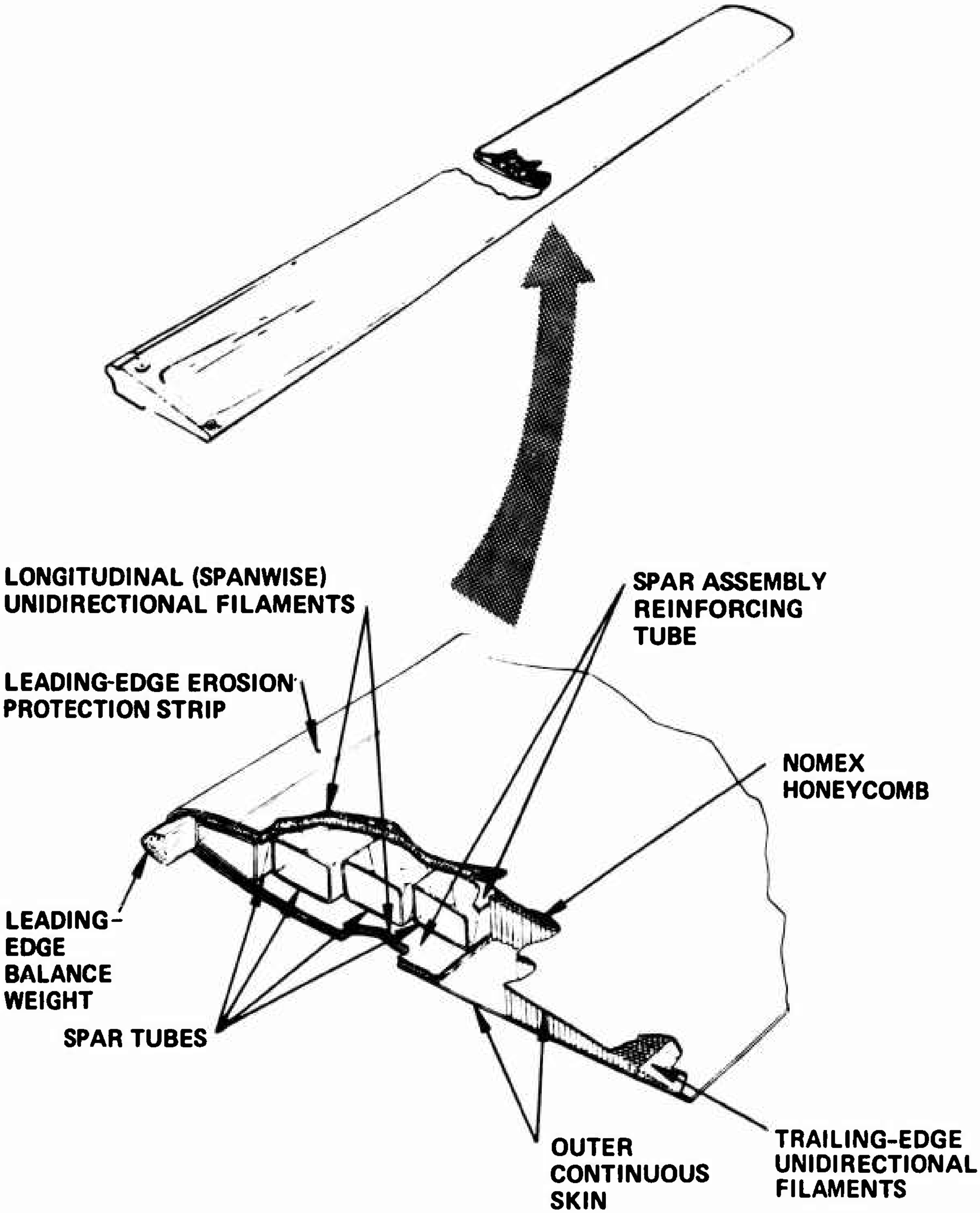


Figure 9. Multi-Tubular Spar Composite Blade.

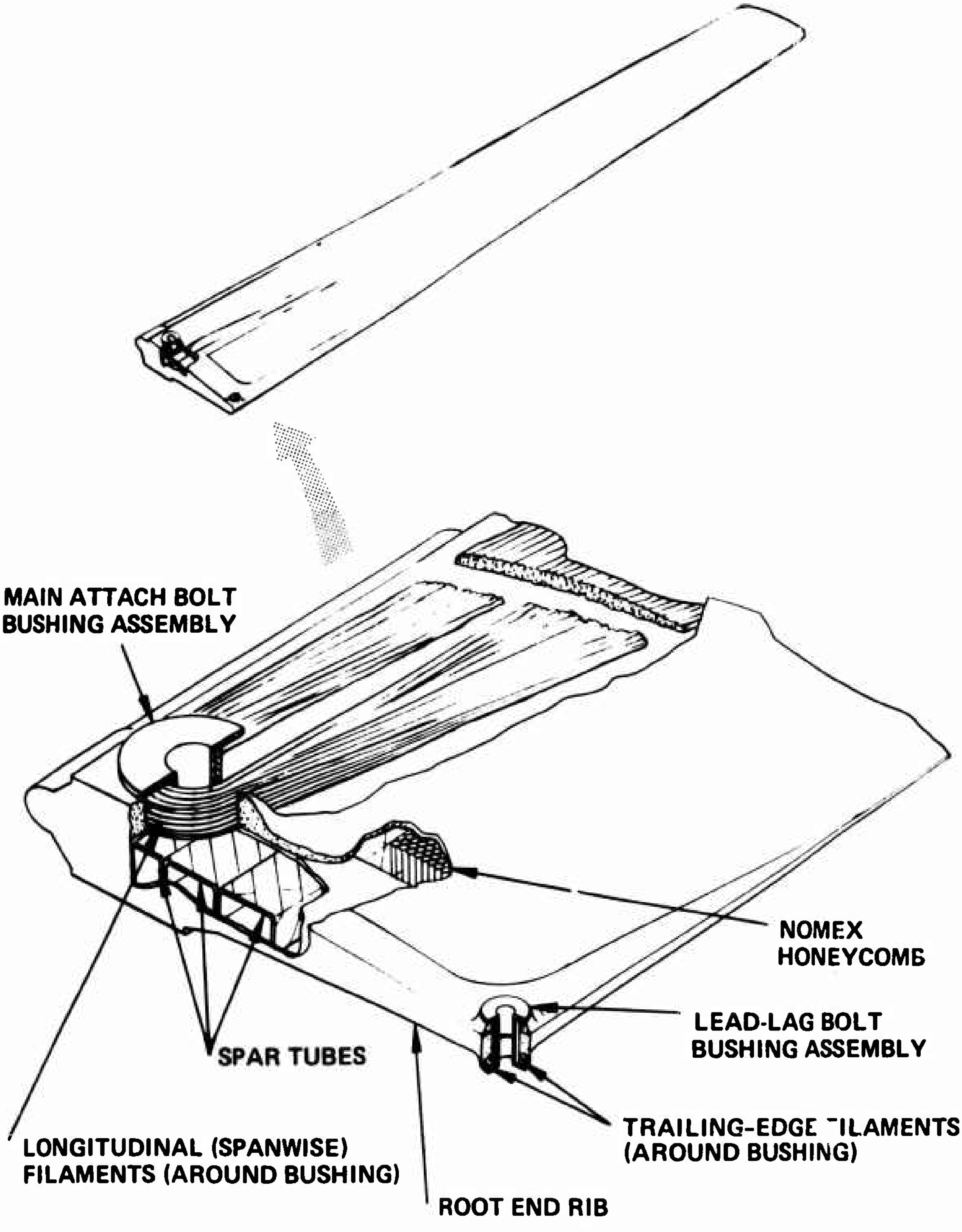


Figure 10. Composite Blade Root End Attachment.

HUB AND BLADE RETENTION SYSTEMS

The various hub and rotor blade retention systems used by current helicopter manufacturing companies have mainly evolved from the structural and mechanical arrangements that they have used successfully in the past. Some of these designs have damage-tolerant features such as multiple wire packs and multiple tension-torsion straps that are side-effect advantages of the original concepts. Some have no alternate load paths or redundancy in case of a failure in a primary member. Fail-safety and redundant load paths have been given increased emphasis in this area in recent years. Also, vulnerability/survivability requirements have focused attention on these items.

Blade folding and blade removal can further complicate the damage-tolerance or redundancy provisions. The basic mechanical simplicity needed to perform these functions in a straightforward manner can be difficult to maintain when redundant members and/or extra mechanical elements are added. Care must be exercised to keep these damage-tolerant provisions from reducing the reliability and maintainability of the primary load-carrying components in this area.

The multiple strap blade retention system (Hughes Helicopters) and the elastomeric blade retention bearings being utilized on many of the new advanced designs have damage-tolerant characteristics by virtue of their basic physical makeup and arrangement.

The OH-6A main rotor strap retention system is an excellent example of a failsafe design. This strap assembly will carry 100 percent of the design centrifugal load after 7 straps out of 15 are broken. Figure 11 shows this two-set strap assembly, which can be used on a four-blade rotor system. Here the centrifugal loads are carried straight through from the lead-lag hinge bolt of one blade to the lead-lag hinge bolt of the opposite blade. The relatively small angle change (kick) loads are transferred into the hub at the bolted connections that transfer the rotor torque to the blades. The hub arrangement enables the strap assembly to be readily inspected by visual means. This design has proven to be extremely rugged and reliable in both civilian and military operations. A total of over 2.5 million hours has been accumulated by Hughes Helicopters using this system.

The latest generation of elastomeric bearings used in new rotor hub designs has the necessary physical makeup for a gradual failsafe mode to occur at the end of the bearing's useful life. The hub design should be configured or have provisions for visual checking for deterioration of the

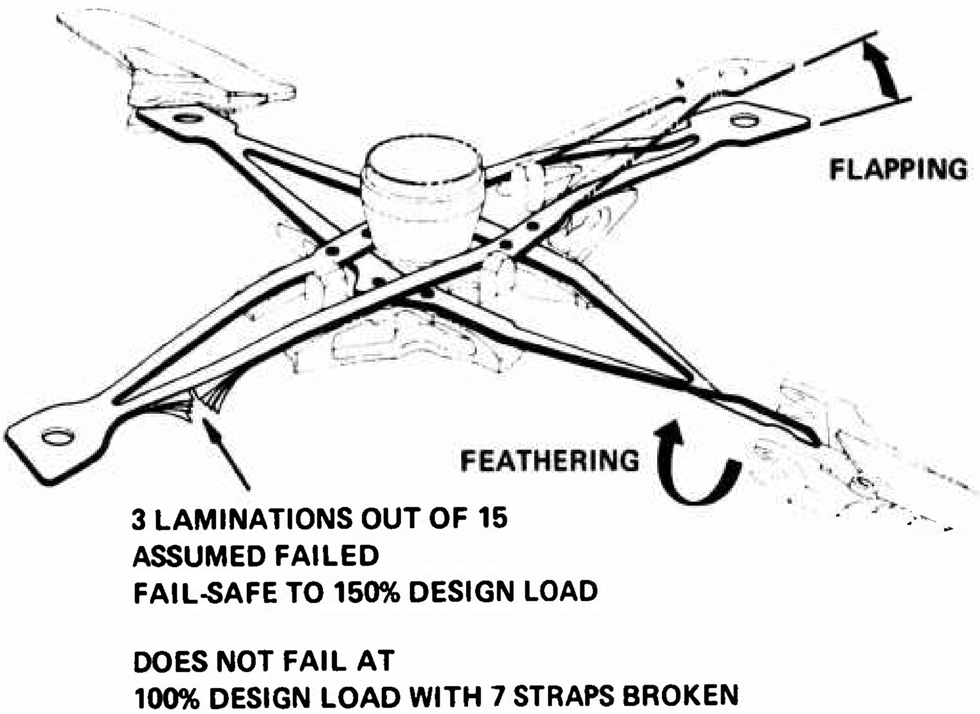


Figure 11. OH-6A Main Rotor Strap Retention.

rubber sections of each bearing. Due to their gradual failure mode, elastomeric bearings can be changed before they reach a dangerous condition. Before the rubber completely fails, there should be a very noticeable change in the spring rate and the friction characteristics of the bearing. The resulting vibration effects on the helicopter due to these changes should be felt by the pilot.

The conventional bearing-retained rotor hubs have various amounts of damage tolerance built into the assembly, which is a function of the types and rates of the individual bearing failure modes. Again, these elements must be inspectable and have some kind of failure indication that can be recognized by operating and maintenance personnel.

After considering the motion provision elements, the remaining hub structure and blade retention components carrying the primary loads must be studied from a damage-tolerance viewpoint. Multiple load paths (redundancy) and the addition of crack-arresting structure are the current state-of-the-art approaches for solutions to this problem.

Care must be exercised with the type and manner of adding redundant structure to achieve the desired fail-safety. Holes for fastening the multi-piece structure together increase stress concentration points which can lower the reliability and fatigue life of the assembly. Also, fabricating a part from multiple pieces can alter the stress distribution in a manner that is detrimental to the overall design. If near-uniform and straightforward load paths can be maintained between the redundant members, the fastening problems may not significantly complicate the design.

One area in which this type of approach should be successful is on a hub assembly like the Army CH-54A TARHE main rotor hub, shown in Figure 12. Here the upper and lower plates could be fabricated by bonding and bolting together several thinner plates, as shown in Figure 13.

Figure 14 is a sketch of a hub assembly similar in size and configuration to the one used on the Hughes Model 269A light helicopter. This is a fully articulated blade and hub rotor system, in which (in this particular arrangement) the flapping hinge is the axis nearest the main rotor shaft (centerline of rotation).

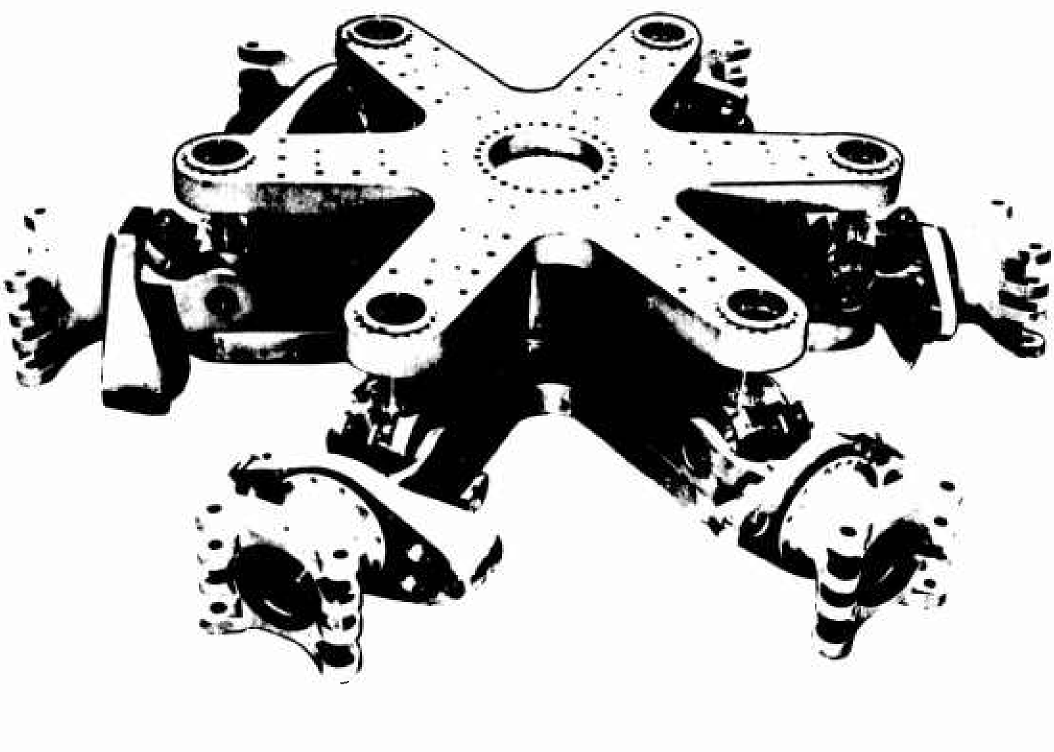


Figure 12. Army CH-54A Hub.

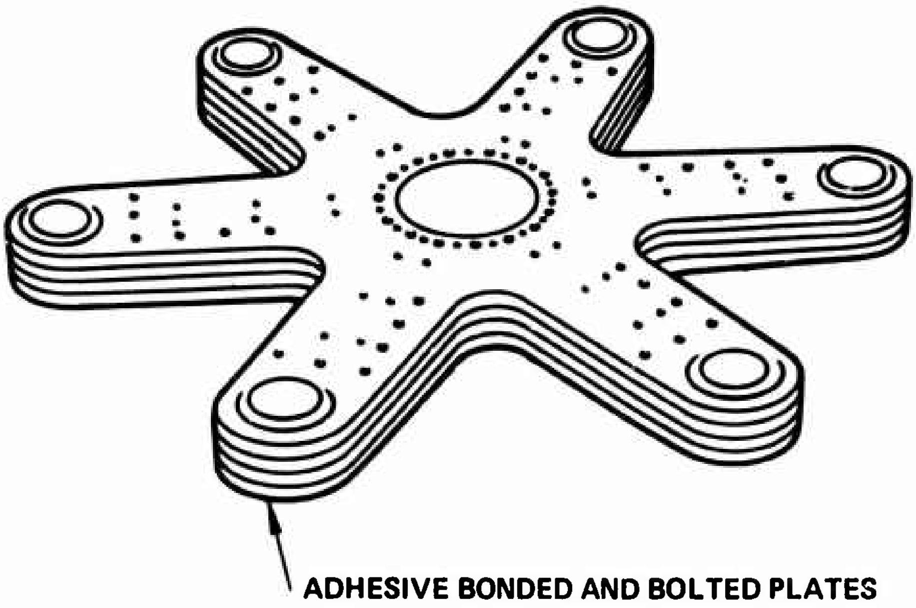


Figure 13. Damage-Tolerant Upper Plate Assembly.

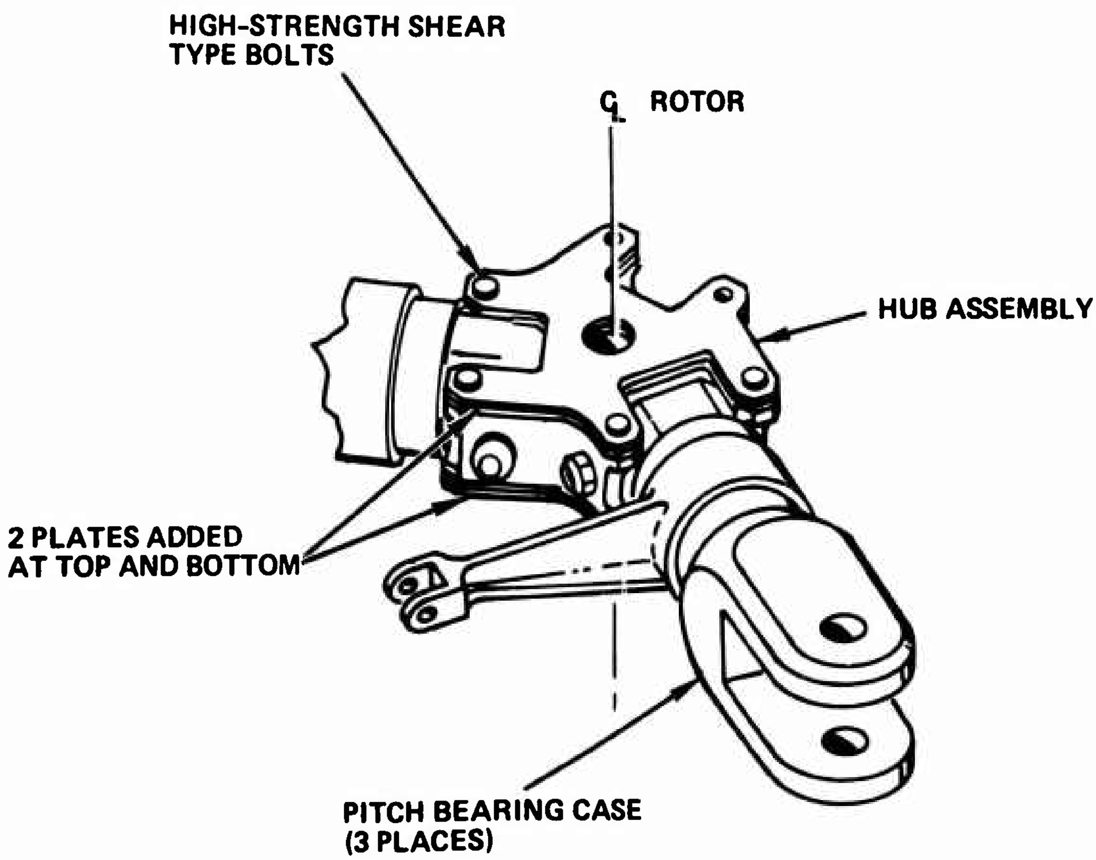


Figure 14. Damage-Tolerant Light Helicopter Hub.

The steel hub configuration shown has been modified by reducing the vertical depth and adding two steel plates on both the top and bottom surfaces. The plates are bonded by adhesives, with a vertical bolt used at each lug end to transmit part of the centrifugal force (CF) loads from the hub to the plates. The major components of these CF loads in the plates would carry across and be balanced out by the other blades.

This study modification was made to add multiple redundancy to a monolithic part of this type. Here the problem is more difficult to accomplish structurally than the stacked plate configuration previously discussed (Figure 13), where the blade lead-lag hinge and flapping hinge are at the same spanwise station.

Adapting the plates as shown will add approximately 40 percent more overall material, which is an added weight of 2.8 pounds (for a total part weight of 9.8 pounds). This involves a cost increase of approximately $280.00 per helicopter.

ROTATING PRIMARY FLIGHT CONTROLS

ROTATING SWASHPLATE

The rotating swashplate, pitch control links, and pitch control arms are critical fatigue-loaded items that need particular attention in addition to that afforded rotor blades and hub and blade retention systems. Wear of the cycling rod-end joints and the possibility of their inducing additional vibrating loads or seizing are important design problems that must be reckoned with. The swashplate bearing is another critical load (force and moment) transfer joint that is absolutely vital to the safety of flight of the helicopter. These items are difficult to make damage-tolerant or redundant without adding significant weight and complication to the control system. Available additional space to accommodate redundant members is usually difficult to find in the area where these rotating controls must be placed. In the vertical direction, efforts are continually being made to keep the distance between the top of the main rotor and the top of the cabin or fuselage to a minimum.

The rotating blade pitch links and pitch control arms generally have close clearances. Adding redundant structure or extra members to achieve fail-safety here conflicts with the continual drive for system simplicity and light weight. However, the total advantages gained by having these components damage tolerant could more than offset the disadvantages over the total life of the helicopter.

Figure 15 is a sketch of a redundant rotating swashplate concept, which is made by bolting two separate parts together. Each of these upper and lower star-shaped plates has an individual clevis at each pitch link attach point. The clevis ears of the upper plate are nested inside those of the lower plate, as shown on the sketch. The through bolts holding the swashplate bearing outer race retaining rings also clamp the two parts together.

The parts would be sized so as to enable flight loads to be carried through several inspection periods when either half is cracked in a critical area. This redundant assembly would weigh approximately 50 percent more (1.5 times) than a comparably designed conventional part. Due to the additional machining and the close tolerances required for fitting the parts together (such as nesting the clevises), the assembly will cost approximately 150 percent more (2.5 times) than the conventional part.

PITCH LINK

Figure 16 is a sketch of a redundant pitch link concept, which is essentially a dual concentric assembly where the outer parts normally carry the flight loads. If there is a failure in the outer load path, the inner assembly picks up the load through the lower compressive modulus inner end bearings. Adequate visual inspection procedures should reveal the failed outer part, since this arrangement should ensure this sequence of failure.

This is a complex concept and is not recommended in the present study form. It will weigh approximately 100 percent more (2 times) and cost approximately 200 percent more (3 times) than a conventional pitch link design.

PROPULSION SYSTEM

DRIVE SHAFT

The main rotor drive shaft shown in Figure 17 is a unique damage-tolerant design in the propulsion system of the Hughes OH-6A helicopter. Because of the floating-axle arrangement of the main rotor hub support, the main rotor drive shaft is not relied upon to transmit primary flight loads to the airframe. In addition, the main gearbox, the airframe, and the main rotor system are designed not to fail prior to the drive shaft in the unlikely event of a lockup in the main rotor gearbox. This example is probably the first instance of a failsafe requirement of this type actually being specified in the detail model specification (Reference 13), thus indicating the customer interest in features of this type. Meeting

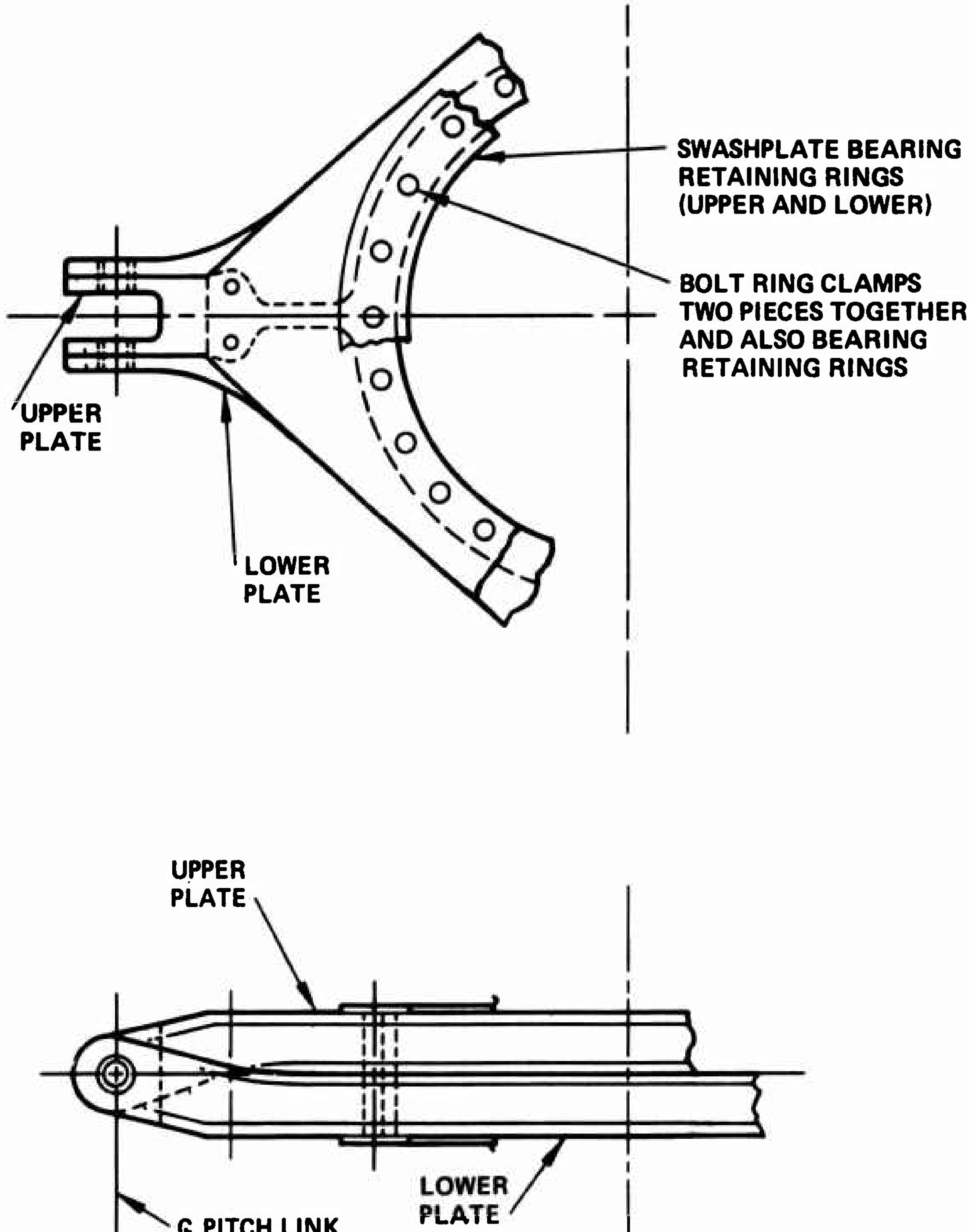


Figure 15. Redundant Rotating Swashplate Concept.

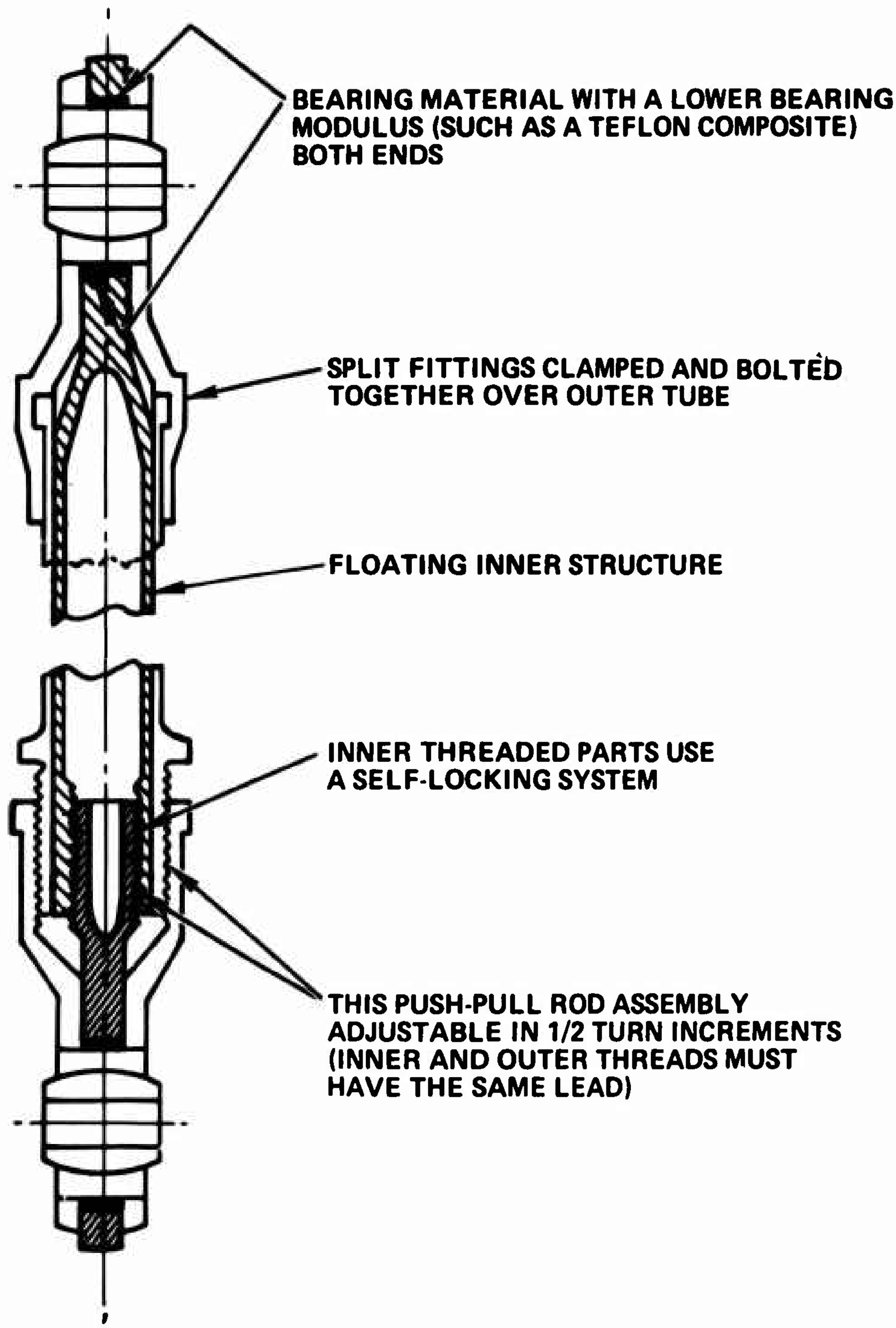


Figure 16. Redundant Pitch Link Concept.

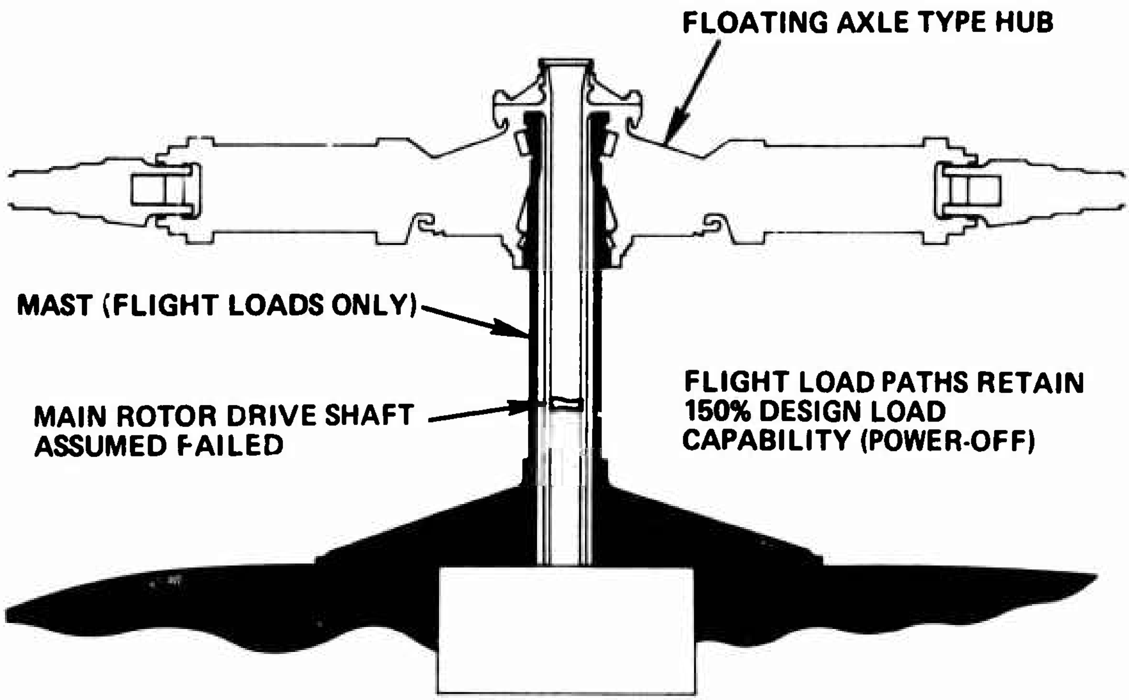


Figure 17. OH-6A Main Rotor Drive Shaft and Floating Axle Type Hub.

this requirement resulted in approximately a 50-percent conservatism over the normal ultimate design torque loads in the affected structural areas. The main rotor drive shaft also carries a service life limitation arrived at by a safe-life approach. This failsafe feature provides an emergency capability of a safe autorotation landing in the event of a structural or mechanical failure anywhere in the main power transmission system.

BEARING INSTALLATION

The bearing installation shown in Figure 18 is an example of a damage-tolerant concept used in the idler pulley installation of the Hughes Model 300 helicopter. The inner bearing race is clamped to the rotating shaft, with the outer bearing race essentially floating in the support bracket. The space between the bearing seals is packed with grease. In the event of the ball bearings' seizing, the outer race would act as a bearing surface, allowing the pulley to continue to function for a limited period.

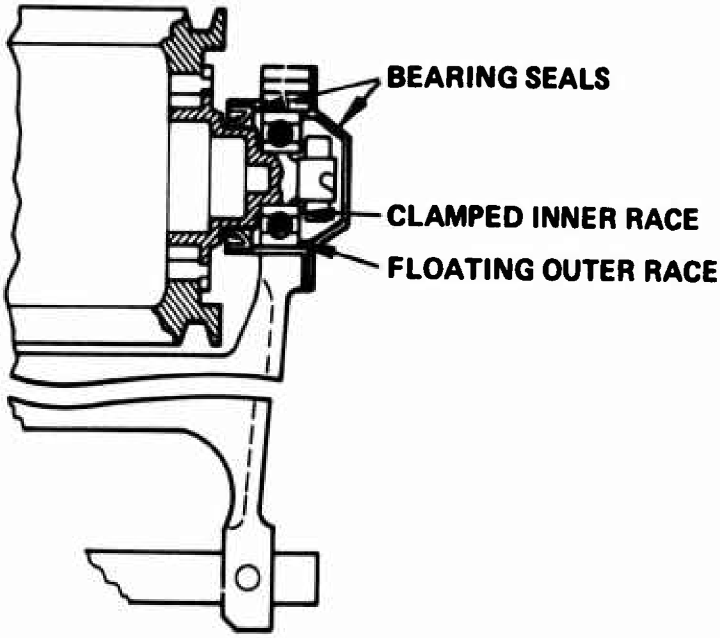


Figure 18. Bearing Installation.

CONTROLLED FRACTURE STRUCTURE

Most of the previous sketches and accompanying discussions have emphasized redundancy as a means of achieving damage-tolerant design. Another very popular and useful design technique is the addition of crack-arresting material in order to get a controlled-fracture type of structure.

In a simplified form, this involves adding doublers or straps of sufficient size to retard the growth of a crack and locally redistribute the applied loads until the failure can be found and the cracked part repaired or replaced. The crack-arrestor/doubler material is adhesively bonded and/or riveted to the primary structure. Various materials are used for this function, with their type and form usually dictated by the requirements of each particular application. The primary structural requirement is that the strap or doubler be of higher strength than the basic material in which it is desired to retard a possible crack.

Fiberglass/epoxy crack-arresting and backup material is being used in some of these specialized applications. Its slow crack-propagation characteristics and its ability to be molded into an intricate shape adjacent to the primary structure are its main attributes in this regard.

Figure 19 (Reference 22) illustrates a typical light frame and stringer type of structure, such as used in pressurized fuselages, that features crack-arrestor doublers bonded between the outer skin and the frame and stringer elements. The curve shows typical test results where a saw-cut crack propagates at a certain rate until it intersects one of the crack-arrestor doublers. At this point, the propagation rate is significantly reduced, and therefore a greater length of time is available for discovering the crack. Visual inspection is primarily relied on to find actual cracks of this nature.

Figure 20 shows another frame crack-stopper configuration for use in fuselage structure (Reference 26). The crack-stopper straps provide continuity across the gap created at the intersection of the fuselage longerons and frames. The stress level in this critical area is reduced to 15 percent below the midbay stress, thus reducing the possibility of a fatigue crack starting. In addition, the crack-stopper strap can be used as frame-bending material to increase the frame stiffness and static strength. Obviously, the straps add weight and cost to the fuselage structure; however, the strap configuration is an effective means of increasing the residual strength of damaged panels and confining unstable fast fracture to local areas.

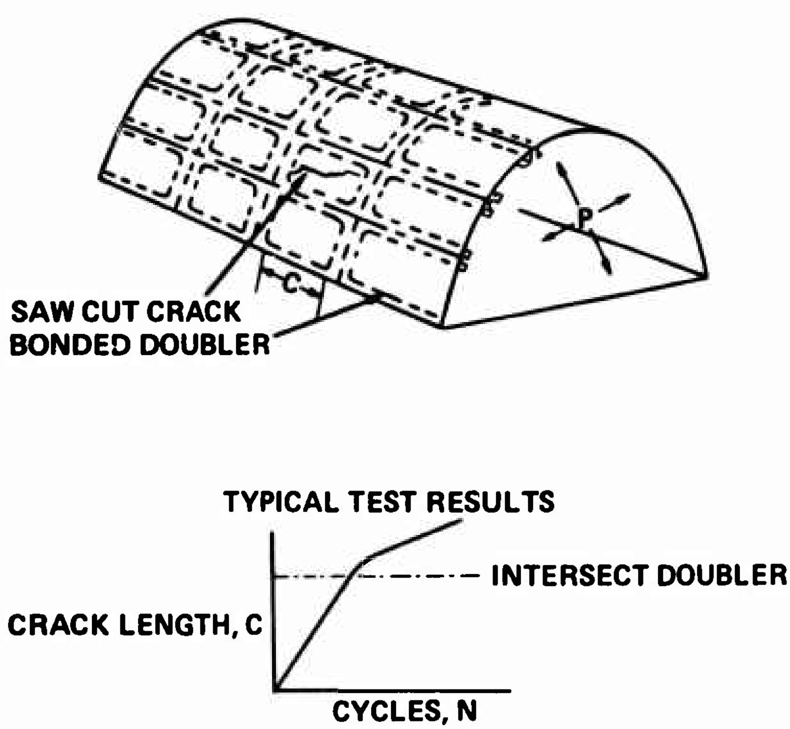


Figure 19. Crack Arrestor Construction Used in Conventional Aircraft Structure.

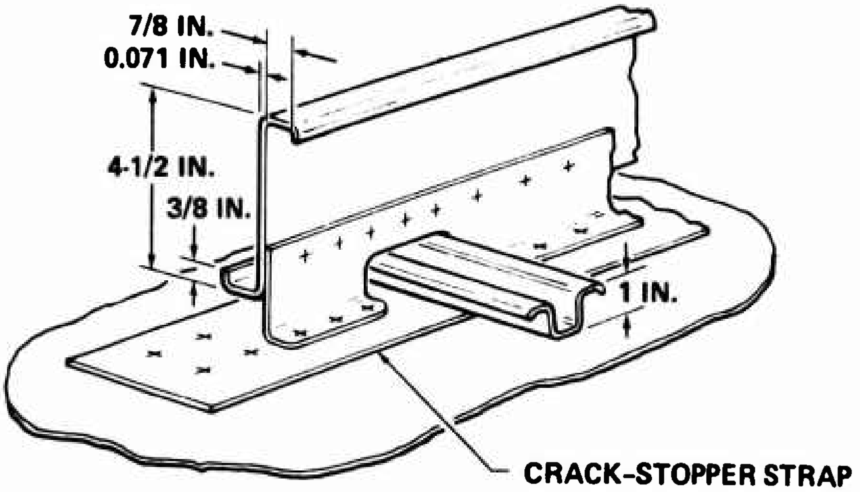


Figure 20. Frame Crack-Stopper Configuration.

FAILURE-INDICATING SYSTEMS

GENERAL

A logically attractive method of coping with the failsafe problem is to incorporate some form of failure-indicating system or systems in the critical safety-of-flight components. This system should be designed and developed to give adequate warning that a potentially catastrophic failure has started to develop. A monitoring procedure or method must be established that is compatible with the modes of failure and will work with the particular warning system.

The extent to which the various possible failure modes are covered and the sensitivity required of the system to be effective determine the sophistication level of the failure warning elements. The effects of an elaborate failure warning system (extremely complicated and sophisticated) can be readily visualized. The weight added would degrade the helicopter performance, and the complexity would greatly increase the cost and maintainability. Therefore, these systems must be carefully selected and integrated into the various components at minimum weight and cost. Some methods and techniques used to accomplish failure detection are discussed below.

ELECTRICAL WIRE SYSTEM

One system that has been tried in the laboratory to a limited extent on rotor blades is the use of multiple electric wires on the outer surface of the main spar or main load-carrying members. These wires are run parallel to the direction of the major loads and are bonded close to the spar surface. Normally, the wires form a closed electrical circuit. The basic function is that a chordwise crack in the main spar or spar assembly will start and, as it progresses, will cause separation of the nearest wire. Therefore, an open circuit is created that can trigger some form of indication or warning to the crew.

This approach has several drawbacks and installation problems. To be most effective, the wires should be placed on the outside of the spar, especially in the areas of high blade bending stresses. It can readily be seen that this creates an airfoil contour (aerodynamic) problem. The wires have to be carefully spaced and bonded to the surface, then covered and faired to the proper aerodynamic contour. Placing the spanwise wires on the inside surface of the spar or spar assembly would reduce their effectiveness against flapwise bending loads. In addition, to accomplish this, inside placement would involve very difficult and costly manufacturing and quality control operations.

Another problem with crack-detection wire applications in general lies in obtaining the correct wire stiffness and strength values that are compatible with the materials being monitored and the adhesives used to bond the wires in place. For an efficient and reliable system, a wire should separate very near the time when it is reached by the crack or separation developing in the spar. The adhesive must be stiff enough in shear to elongate or compress the wire along with the movement of the spar surface (i. e., minimum relative motion is desired between the spar and each individual wire). However, a brittle adhesive will tend toward fatigue failures, causing subsequent problems.

To get a warning signal to the crew station requires a slip-ring assembly somewhere along the centerline of the main rotor shaft or hub assembly. In addition, the required electronic black boxes and wiring assemblies further complicate the overall installation.

As has been pointed out in this brief discussion, the electrical wire system is complicated and costly and has not been generally accepted on actual helicopter components.

ELECTRICAL CAPACITOR SYSTEM

An electrical capacitance type of system has been considered by one helicopter manufacturer. Here alternating thin layers of insulating and conductive materials are applied to the spar surfaces or to whatever surface is being monitored. The capacitor cracks and, therefore, discharges when the spar is cracked. This discharge is transmitted and recorded in some manner in the aircraft.

The problems and difficulties of carefully applying the thin layers of material would be very costly. A slip-ring assembly and other electronic components would also be required, as in the electrical crack wire detection system. Due to these problems and high costs, this system has not been further pursued.

PRESSURE LEAK DETECTION SYSTEMS

A successful method that has been developed for use on rotor blades by two manufacturers of large-size helicopters in the United States is the use of differential pressure to indicate the structural integrity of the major load-carrying spars. Sikorsky Aircraft Division of United Aircraft Corporation uses its Blade Inspection Method (BIM) system. The Vertol Division of the Boeing Company uses its Integral Spar Inspection System (ISIS).

The Sikorsky BIM system uses a positive pressure on the inside of the main spar tube. The inboard and outboard ends of the main rotor blade spar tube are sealed, forming a long spanwise air chamber. A pressure indicator is mounted at the inboard end of the spar extrusion or tube, where it can readily be inspected while the helicopter is on the ground. A crack or leak will allow the inside pressure to equalize with ambient pressure, which will be shown on the pressure indicator at the next ground inspection.

One disadvantage of this system is that centrifugal force will vary the pressure spanwise along the blade spar. This could cause airfoil contour deformation problems at the tip, where the contour is most critical, requiring additional pressure reinforcing structure.

The Boeing-Vertol ISIS system utilizes a vacuum liner assembly with reduced pressure between the internally installed liner assembly and the inside of the spar tube. This feature eliminates the major effects of the centrifugal force problems noted above in the pressurized system. The liner is a Mylar-aluminum layered composite with a Dacron bleeder cloth on the surface next to the spar. At the tip end of the blade, a rib or bulkhead is sealed off and provides the end support for the liner. A visual indicator and sensor assembly are located at the root end. The volume of reduced-pressure air is quite small in the system, primarily consisting of the circumferential layer within the bleeder cloth.

Both of the above systems only monitor the structural integrity of D-section or similar spars. A multiple-cell spar assembly could utilize the BIM (pressure) system; however, the ISIS (vacuum with liner) system would be difficult and heavy to use. Also, there are problems with leading-edge erosion protection strips and de-icing blankets bonded to the spar that seal off this part of the spar surface. Any corrosion-protective primers or paints should have crack-propagation characteristics very similar to those of the material being monitored. A tough elastic-type coating could tend to keep the spar differential pressure sealed until the crack became excessively large.

These pressure leak detection systems do not monitor the many other critical parts of the blade, such as the root end fittings, doubler debonding, etc. To be effective, they must be used where the spar is definitely proven by extensive tests to be the first element where a crack begins (i.e., the spar is the weak fatigue link in the structural system). The root end fittings and lugs must continue to be visually monitored to cover any cracks from material flaws and scratches that would cause failures to occur in these different areas.

Of the two leak detection systems, the BIM arrangement should be the lightest and least expensive, provided that the centrifugal pressure problem can be overcome without excessive complications. Having a spanwise pressure gradient while the blade is rotating should not be a handicap. The indicator on the root end of the blade can be inspected only when the blade is stationary. Here, the pressure will be back to the constant spanwise value. On the ISIS arrangement, providing and installing the liner assembly is the greatest drawback.

Both systems should be cost-effective if the failure modes can be reliably established as some form of crack starting in the spars. Increasing the reliability and prolonging the service lives of expensive components, such as rotor blades, justify systems of these types.

MONOLITHIC STRUCTURE WITH ZERO CRACK GROWTH

Monolithic structure with zero crack growth is defined as a single load path structure designed for a safe life. Structural characteristics included in the design to ensure an adequate safe life are:

1. Materials having good fatigue and low-crack-propagation characteristics are selected.
2. Processing and fabricating techniques that have little or no detrimental effects on fatigue are used.
3. The structure is sized to produce very low operational stress levels.

Monolithic structure generally tends to be simpler and less expensive to manufacture than structure with redundancy and/or controlled fracture techniques. Unacceptable weight penalties will usually be incurred in maintaining the low operating stress levels necessary to prevent crack growth unless:

1. The component is small

or

2. Minimum fabricating thicknesses result in low operating stresses.

Examples of monolithic structure that can be cost and weight effective are control rods and pitch links. The maximum design loads for these components are high relative to the operating loads. This results in low operating stress levels. The components are of small size and are normally located in areas allowing ease of inspection.

All critical flight structures are exposed to numerous kinds of damage during the operational life of the helicopter. Once damaged, stress concentration points are formed, and cracks will eventually propagate under repeated loadings. Unless these cracks are found and repaired or the component is replaced, a failure is likely to occur. Monolithic structure has minimal damage-tolerant characteristics and is more dependent on inspection techniques and frequency of inspection intervals to locate damages or cracks to prevent catastrophic failures.

COST AND WEIGHT TRADE-OFF STUDY

INTRODUCTION

The Hughes OH-6A main rotor blade was selected as the structural component to establish the effects on costs and weights in designing to different structural criteria. The study also includes the effects of using different damage-tolerant design techniques. The structural criteria include static strength, fatigue strength (safe-life) and damage-tolerant strength. The damage-tolerant design techniques selected for the study are controlled fracture structure, redundant structure, and combined metal and composite structure.

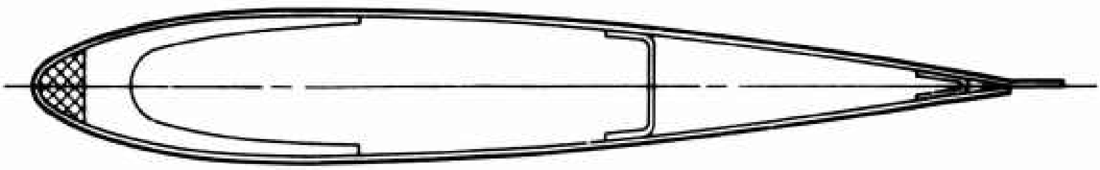
CONCEPT A - HUGHES OH-6A MAIN ROTOR BLADE

The OH-6A main rotor blade is a quasi-monolithic structure with the primary load-carrying elements consisting of an aluminum leading-edge spar and a continuous wraparound aluminum skin. A cross section of the blade is shown in Figure 21 as concept A. The simple blade structure is easy to manufacture and produces a smooth, protuberance-free aerodynamic surface required for the high-performance OH-6A helicopter.

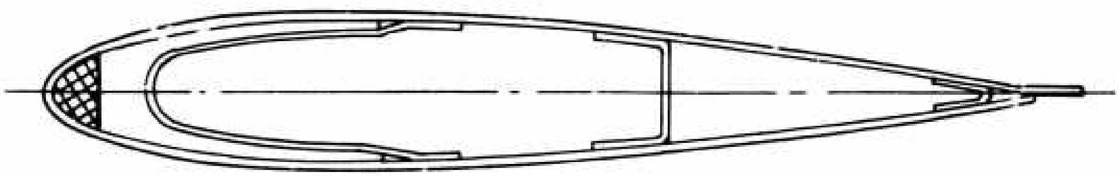
The structural criteria existing when the blade was designed consisted of static strength requirements and fatigue strength requirements including the establishment of a fatigue life (safe-life). No damage-tolerant (failsafe) criteria existed when the OH-6A blade was designed. However, the damage-tolerant techniques recognized at that time were incorporated in the blade design. One of these was to require the fatigue strength of the spar material to be higher than the skin material. The spar is covered by the skin and is not readily inspectable, whereas the skin is inspectable. The blade skin is subjected to higher combined stresses; therefore, if a blade crack should develop, it would occur in the skin. Numerous blade fatigue tests have confirmed this mode of failure. The attachment of the root fittings to the rotor blade is another damage-tolerant (failsafe) technique incorporated in the design of the OH-6A blade. The root fittings are both bonded and bolted to the blade. The primary load transfer from the blade to the fittings is through the bond line. The bolts provide two structural capabilities:

a. They prevent prying loads on the bond line.

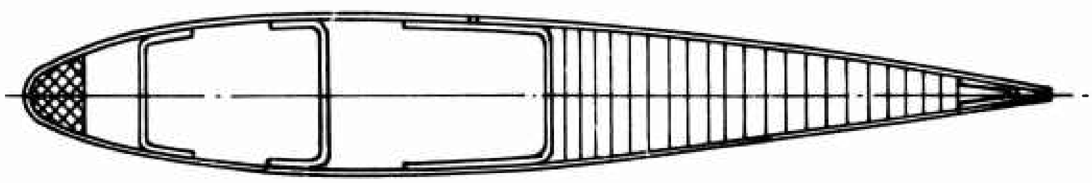
b. They are structurally capable of sustaining ultimate loads should the bond line fail.



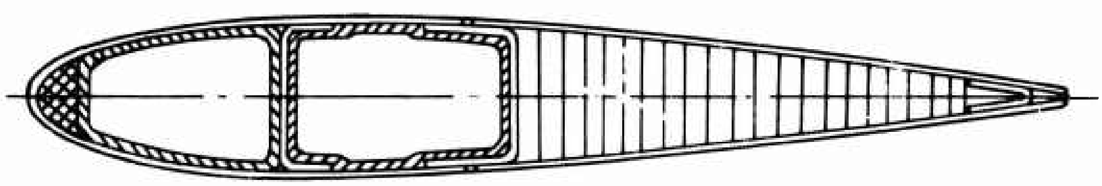
CONCEPT A – OH-6A ALUMINUM BLADE



CONCEPT B – ALUMINUM CONTROLLED FRACTURE BLADE



CONCEPT C – STEEL REDUNDANT BLADE



CONCEPT D – METAL AND COMPOSITE BLADE

Figure 21. Blade Structural Concepts.

The OH-6A blade has given excellent service under all normal types of usage and environments. In combat conditions, there are numerous accounts on record verifying the survivability of the blade for short periods of time when subjected to ballistic or tree-strike damage. The prototype OH-6A blades were substantiated for a fatigue life (service life) of 1655 hours. Detail changes were made in the construction of the production OH-6A blades to provide additional improvement in fatigue strength and in maintaining the airfoil shape. These changes resulted in an increase in fatigue life to 2450 hours. Because the OH-6A main rotor blade is a quasi-monolithic structure, the degree of allowable damage in terms of scratches, nicks, gouges, or dents is minimal with little or no rework possible. This has required blades to be replaced prior to the blades' reaching their established safe-life and during the time period from 1967 to 1970 resulted in an average replacement or failure rate of 650 hours.

CONCEPT B - CONTROLLED FRACTURE STRUCTURE

The construction of the OH-6A main rotor blade is revised to include controlled fracture design technique. The blade revisions consist of the following:

a. The addition of a formed sheet metal channel that is bonded to the inside of the main C-spar extrusion and extends aft to bond to the blade skin.

b. Increasing the thickness and material strength of the rear spar channel.

The purpose of the added C-spar channel is to reduce the stress concentration at the edge of the spar and to act as a crack arrestor to possible cracks developing in the skin at the point of maximum stress. The increase in strength of the rear spar channel is to provide a crack arrestor to a possible chordwise crack originating from the blade trailing edge. A cross section of the controlled fracture blade concept is shown in Figure 21.

The structural revisions to the OH-6A blade to provide crack arrestors would provide little or no increase in the fatigue life of the blade. The structural improvement of the blade would be the increase in the degree of damage, in terms of scratches, dings, gouges, or dents, that the blade could sustain and continue to be operational. The estimated average replacement or failure rate is 1100 hours as compared to the present 650 hours experienced on a current OH-6A blade.

CONCEPT C - REDUNDANT BLADE STRUCTURE

The redundant blade damage-tolerant concept consists of multiple spars made from thin, high-strength stainless steel with slow crack propagation characteristics. A material with these structural characteristics is AM355. The spars are the primary load-carrying structure, with the blade aft section a composite structure consisting of S-glass outer skin and nomex honeycomb core. The OH-6A blade design parameters were used to establish the multi-spar design and to make comparison to the OH-6A blade as compatible as possible. A cross section of the steel redundant blade concept is shown in Figure 21.

Stainless steel is less susceptible to damages such as scratches, dings, gouges, or dents than the aluminum skin of the present OH-6A blade. The redundant spar construction would increase the allowable degree of damage before requiring the blade to be replaced and should meet or exceed the recommended damage-tolerant criteria. The estimated average replacement or failure rate is 3500 hours. However, the cost to manufacture the steel blade concept would be higher than the present OH-6A blade. The material cost of AM355 stainless steel is 10 times higher than the cost of aluminum sheet. Stainless steel is more difficult to fabricate than aluminum. The steel spar blade structure requires an increase in the number of parts to be manufactured, processed, inspected, and assembled. The estimated production cost to manufacture the steel redundant blade structure is $2800.

CONCEPT D - METAL AND COMPOSITE STRUCTURE

The metal and composite blade structure is similar to the previously discussed steel redundant blade, concept C, except that it has one less steel spar and the addition of two composite tubes. A cross section of the blade is shown in Figure 21. The composite tubes would be essentially unloaded during the operation of an undamaged blade due to the differences in modulus of elasticity. However, should the steel spars sustain damage, the composite tubes would provide load redundancy and act as crack arrestors for the steel spars. This concept has desirable structural features in the event of ballistic damage. Metals tend to tear when impacted by projectiles, and the effect of the composite tubes would be to limit this tearing action.

The metal and composite blade should be capable of safety sustaining more damage than the steel redundant blade, concept C; however, the cost to manufacture would be higher. The estimated average replacement or failure rate is 5000 hours, and the manufacturing cost is $3000.

COST ANALYSIS

The first objective is to develop a cost model that relates the effects of different structural design criteria. The following areas are affected:

a. Design

b. Formal substantiation

c. Manufacture

A major item is that group of fixed costs which is incurred independent of the quantity of the components procured. These costs are represented by the total cost required to design, analyze, and test. A second major item is that which is variably dependent upon the quantity of the components produced or procured. These costs are represented by the cost to manufacture the component and the number of components required per flight hour.

The total cost equation for a structural component is as follows:

$$C_T = C_{DS} + nC_M F_H$$

where

C_T = Total cost

C_{DS} = Cost to design and substantiate

n = Number of components required per flight hour

C_M = Cost to manufacture per component

F_H = Flight hours

The cost equation is now derived for determining the costs of designing, substantiating, and manufacturing new blades incorporating damage-tolerant features relative to the cost of the present OH-6A main rotor blade. The cost equation is as follows:

$$C_T = R_{DC}\, C_{DO} + 4\,(1/F_R)\, R_L\, R_{MC}\, C_{OH}\, F_H$$

where

C_{DO} = Cost to design and substantiate the OH-6A blade

R_{DC} = Complexity ratio to design and substantiate the damage-tolerant blade concepts relative to the OH-6A blade

4 = Number of blades per helicopter

F_R = Failure rate - OH-6A blade

R_L = Inverse ratio of expected field service life relative to OH-6A blade

R_{MC} = Complexity ratio to manufacture damage-tolerant blade concepts relative to cost to manufacture OH-6A blade

C_{OH} = Cost of OH-6A blade

The costs to design and substantiate the OH-6A main rotor blade to static and fatigue structural criteria are presented in Table 4. These costs are based on Hughes Helicopters records, with the rates to convert man-hours to dollars based on 1974 rates. The costs to design and substantiate the OH-6A blade to the recommended damage-tolerant criteria are best estimates and are included in the table. The total costs to design and substantiate a blade to static, fatigue, and/or damage-tolerant criteria are presented in Table 5.

The factors involved in determining the total cost of the damage-tolerant blade concepts for comparison to the present OH-6A blade are presented in Table 6. Also included in the table are the relative blade weights for the different designs. The cost equations are derived for the OH-6A blade and for the damage-tolerant blade concepts and presented on page 82.

The total blade costs are computed versus total flight hours. The total cost is then divided by the total flight hours to give the average cost per flight hour versus total flight hours. The results are presented in Table 7. The total flight hours can represent either the flight time of a single helicopter or the total time of a fleet of helicopters. All factors involved in a complete life-cycle cost analysis have not been included in this cost analysis. Therefore, the cost per flight hour is normalized relative to the cost per flight hour of the OH-6A blade for a total of 1000 flight hours.

TABLE 4. OH-6A BLADE - COSTS TO DESIGN AND SUBSTANTIATE[1]

Group - Criteria	Man-Hours	Average Rate Per Hour[2]	Cost
DESIGN		$23.13	
a. Static	900		$20,820
b. Fatigue	1000		23,130
c. Damage Tolerant	1100		25,440
ANALYSES		26.63	
a. Static	420		11,180
b. Fatigue	500		13,320
c. Damage Tolerant (per failure mode)	900		23,970
STRUCTURAL TESTING		15.31	
a. Static	60		920
b. Fatigue (per blade)	640		9,800
c. Damage Tolerant (per blade per failure mode)	900		13,780

NOTES:

(1) The costs to design and substantiate to static and fatigue criteria are based on Hughes Helicopters records. The costs to design and substantiate the OH-6A blade to the recommended damage-tolerant criteria are best estimates.

(2) The average rates per hour are 1974 rates.

TABLE 5. TOTAL DESIGN AND SUBSTANTIATION COSTS			
Criteria	Static	Fatigue(1)	Damage Tolerant(2)
Design	$20,820	$23,130	$25,440
Analyses	11,180	13,320	71,910
Test	920	38,200	82,680
Total	32,920	75,650	180,030
Static only	32,920	-	-
Static + Fatigue	-	108,570	-
Static + Fatigue + D. T.	-	-	288,600

NOTES:

(1) A minimum of four blade tests are required to establish fatigue strength.

(2) No requirement exists for the amount of testing and analyses necessary to prove compliance with damage-tolerant criteria. The probable minimum requirements are two blades tested for each mode of failure with three critical failure modes tested, resulting in a total of six blades to be tested. This would also require a minimum of three independent fracture analyses in conjunction with the test results.

TABLE 6. FACTORS FOR BLADE COST EQUATIONS AND RELATIVE BLADE WEIGHTS[1]

Concept	Description of Blade Concept	R_{DC}	Cost to Manufacture	R_{MC}	Failure Rate	R_L	Blade Weight (pounds)	Percent Weight Increase
A	page 72	1.00	$1670	1.00	650	1.00	27.0	0
B	page 74	1.10	1850	1.10	1100	0.60	28.6	6
C	page 75	1.30	2800	1.67	3500	0.19	29.4	9
D	page 75	1.40	3000	1.80	5000	0.13	29.2	8

NOTE:

(1) For definition of R_{DC}, R_{MC}, F_R, and R_L, refer to page 77.

For better visibility, curves of the relative cost per flight hour versus total flight hours for the different design concepts are plotted and shown in Figure 22.

TOTAL COST EQUATIONS

The total cost equations are now obtained for each of the blade design concepts from the basic equation:

$$C_T = R_{DC}\ C_{DO} + 4\ (1/F_R)\ R_L\ R_{MC}\ C_{OH}\ F_H$$ (Ref page 77)

Blade Concept	Total Cost Equation
A	$C_{TA} = 108{,}570 + 10.277\ F_H$
B	$C_{TB} = 317{,}460 + 6.783\ F_H$
C	$C_{TC} = 375{,}180 + 3.261\ F_H$
D	$C_{TD} = 404{,}040 + 2.405\ F_H$

where

F_H = Total flight hours

TABLE 7. BLADE TOTAL COST, COST PER FLIGHT HOUR, AND RELATIVE COST PER FLIGHT HOUR VERSUS TOTAL FLIGHT HOURS

	CONCEPT A			CONCEPT B		
Total Flight Hours	Total Cost	Cost Per Flight Hour	Relative Cost Per Flight Hour	Total Cost	Cost Per Flight Hour	Relative Cost Per Flight Hour
1000	\$118,800	\$111.80	\$1.000	\$324,200	\$324.20	\$2.728
10000	211,300	21.13	0.178	385,300	38.53	0.324
100000	1.14×10^6	11.40	0.097	996,000	9.96	0.084
1×10^6	1.03×10^6	10.30	0.086	7.10×10^6	7.10	0.060
	CONCEPT C			CONCEPT D		
Total Flight Hours	Total Cost	Cost Per Flight Hour	Relative Cost Per Flight Hour	Total Cost	Cost Per Flight Hour	Relative Cost Per Flight Hour
1000	\$378,400	\$378.40	\$3.184	\$406,400	\$406.40	\$3.419
10000	407,800	40.78	0.343	428,100	42.81	0.360
100000	701,300	7.01	0.059	644,500	6.45	0.054
1×10^6	3.64×10^6	3.64	0.031	2.81×10^6	2.81	0.024

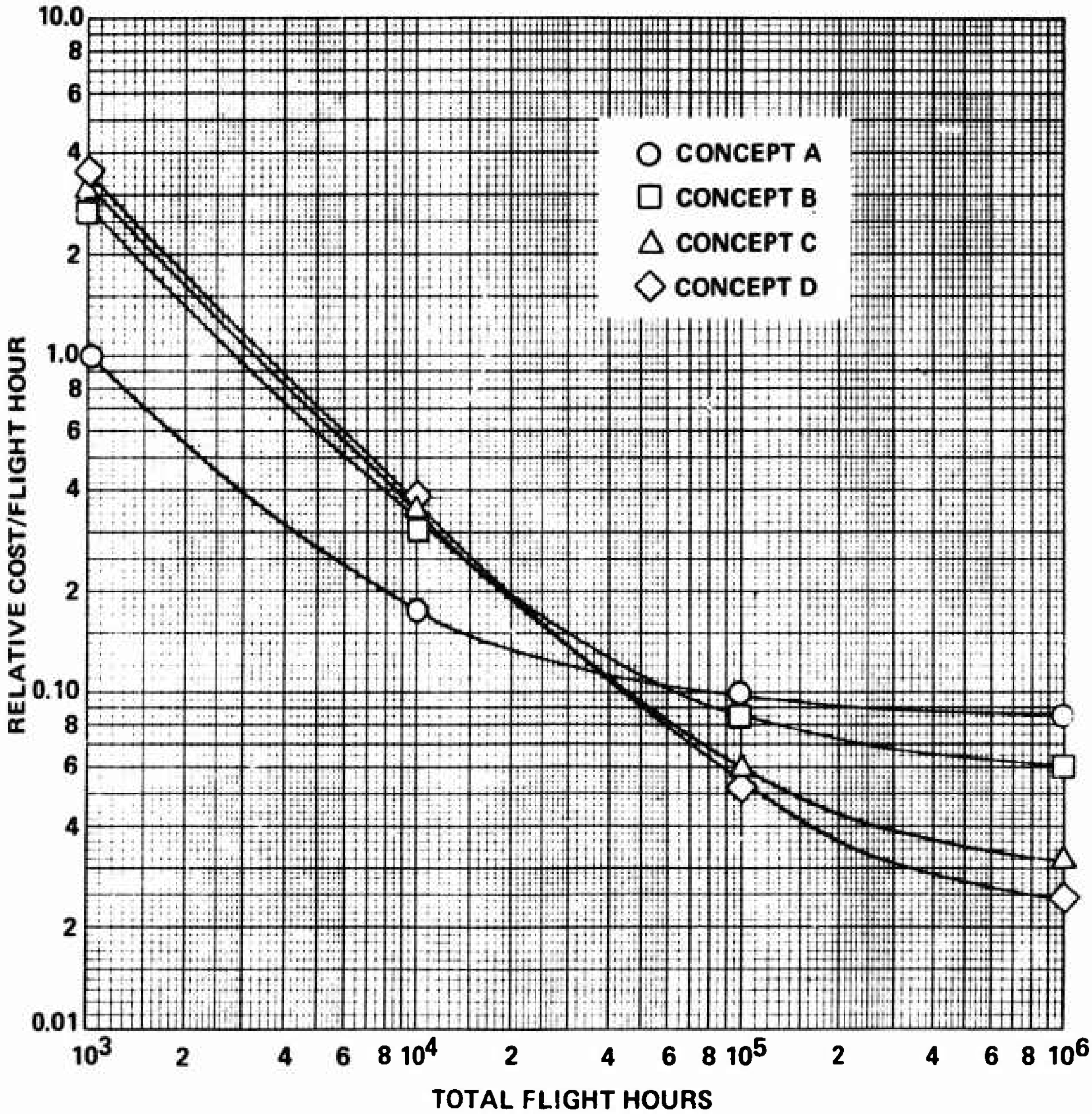


Figure 22. Blade Concept Cost Comparison.

CONCLUSIONS

Primary conclusions from the failsafe/safe-life interface criteria study are as follows:

a. "Damaged" strength of structural components requires increased emphasis.

b. Inclusion of damage-tolerant concepts should reduce requirements for formal substantiation of safe-lives.

c. Acceptable inspection methods should permit the use of components beyond their computed safe-life.

d. Inspection of critical structural areas should be made easier.

e. "Damage-tolerant" should replace "failsafe" as a descriptive term.

This study showed the following to require definition:

a. Types and number of failure modes to be considered.

b. Failure modes to be substantiated by test and/or analysis.

c. Acceptable methodology for establishing inspection periods.

The most costly item in designing to damage-tolerant (failsafe) criteria is to show formal compliance of the helicopter structural components with the criteria. The results of this study clearly indicate that, unless an economically feasible method of substantiating damage-tolerant structure is approved by both Government agencies and helicopter manufacturers, the manufacturers will continue to request deviations from specifications and will design to the concept of a computed safe-life and a failsafe criterion that does not involve the establishment of inspection periods.

There are several design techniques to improve the damage tolerance of structural components. These techniques all involve an initial increase in cost and weight, which can be minimized with careful attention to detail design. It is expected that helicopter components manufactured to include damage-tolerant concepts will survive longer in service, thereby resulting in a lower life-cycle cost.

RECOMMENDATIONS

1. It is recommended that the section on "Damage-Tolerant Design Criteria" (pages 43 through 47) be circulated to both industry and Government agencies for their opinions and comments. Areas of primary concern are as follows:

    a. Economically feasible methods to formally substantiate that structural components comply with the damage-tolerant criteria.

    b. Reliability of field inspection.

    c. Size of damage or crack probable of detection.

    d. Size of damage or crack relative to the component size, required in determining the time of possible failure.

    e. Frequency of inspections relative to the established time interval from probable detection to possible failure.

2. It is recommended that additional cost and weight trade-off studies be performed for other primary structural components to better determine the weight penalties for incorporating damage-tolerant concepts. The OH-6A blade used for the cost and weight trade-off study was a good selection for determining the cost to design different structural criteria using various damage-tolerant concepts. However, blade weight is a critical blade design parameter, and the study does not fully reflect the weight penalties involved in incorporating damage-tolerant concepts.

3. A study is recommended to establish probable structural failure modes by reviewing the service experience of various types of helicopters. Emphasis should be placed on types of structural damage that resulted in a catastrophic failure.

4. A study is recommended to determine the reliability of fracture analysis methodology when applied to structural components. The study would consist of analyzing and testing small-scale specimens that simulate actual helicopter structural components. The test results would be compared to the fracture analysis for verification or recommendations for improvements.

5. A study is recommended of the latest state-of-the-art inspection procedures, techniques, reliability, etc., with emphasis on field service inspection.

LIST OF SYMBOLS

C_{DO} = Cost to design and substantiate the OH-6A blade

C_{DS} = Cost to design and substantiate

C_M = Cost to manufacture per component

C_{OH} = Cost of OH-6A blade

C_T = Total cost

F_H = Flight hours

F_R = Failure rate - OH-6A blade

n = Number of components required per flight hour

R_{DC} = Complexity ratio to design and substantiate the damage-tolerant blade concepts relative to the OH-6A blade

R_L = Inverse ratio of expected field service life relative to OH-6A blade

R_{MC} = Complexity ratio to manufacture damage-tolerant blade concepts relative to cost to manufacture OH-6A blade